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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

APOLLO 16

30-DAY FAILURE AND ANOMALY LISTING REPORT

MANNED SPACECRAFT CENTER

HOUSTON, TEXAS

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1.0 INTRODUCTION

This report contains a discussion of the significant anomalies that occurred during the Apollo 16 mission. The discussion is divided into five major sections: command and service modules, lunar module, government-furnished equipment, lunar surface experiments, and orbital experiments. All times are elapsed time from range zero, established as the integral second before lift-off.

2.0 COMMAND AND SERVICE MODULE ANOMALIES

2.1 WATER/GLYCOL TEMPERATURE CONTROL CIRCUIT FAILED IN THE AUTOMATIC MODE

About 3 hours into the mission, the water/glycol temperature control circuit (fig. 2-1) malfunctioned in the automatic mode. The malfunction allowed an excessive amount of hot water/glycol to bypass the radiator causing the mixed water/glycol temperature to exceed the upper control limit of 48° F. After remaining in this maximum bypass position for 5 minutes, the temperature control valve cycled regularly between maximum and minimum flow to the radiators for about 14 minutes before stopping again at maximum bypass. The control was changed to manual and back to automatic to restart the valve operation and, finally, left in manual at the desired valve position. Variations in the evaporator outlet temperature and total system flow rate for the period are shown in figure 2-2. Control valve operation is evident from the variation of system flow rate. Total flow rate increases as the flow control valve opens since the total system pressure drop decreases.

The temperature control valve was subsequently positioned manually several times to maintain acceptable coolant loop temperatures under various thermal conditions. During two attempts to position the valve automatically (before and after the transearth extravehicular activity) the valve appeared to position correctly but, only for a short time.

Most likely, a malfunction occurred in the water/glycol temperature controller which modulates the control valve by supplying signal pulses with a duration proportional to the error of the mixed temperature sensed.

Postflight testing of the system is scheduled.

This anomaly is open.

2.2 SERVICE PROPULSION SYSTEM OXIDIZER TANK PRESSURE MEASUREMENT SHIFTED

The service propulsion system oxidizer tank pressure measurement shifted upward 15 psia after the spacecraft reached a vacuum environment. The measurement responded to normal changes in pressure throughout the mission, but always read 15 psia high as verified by other measurements in the system.

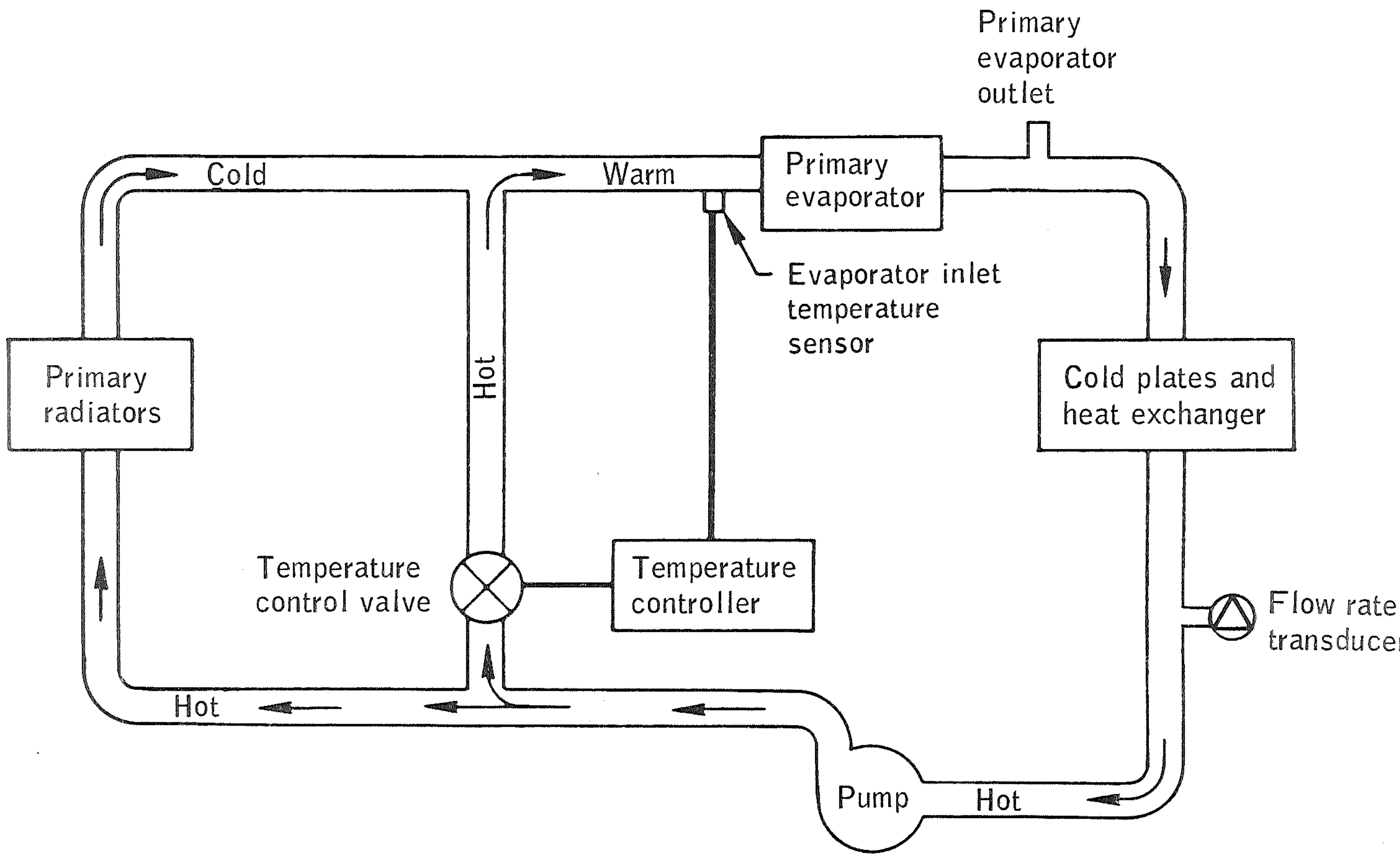


Figure 2-1.- Primary water/glycol coolant loop.

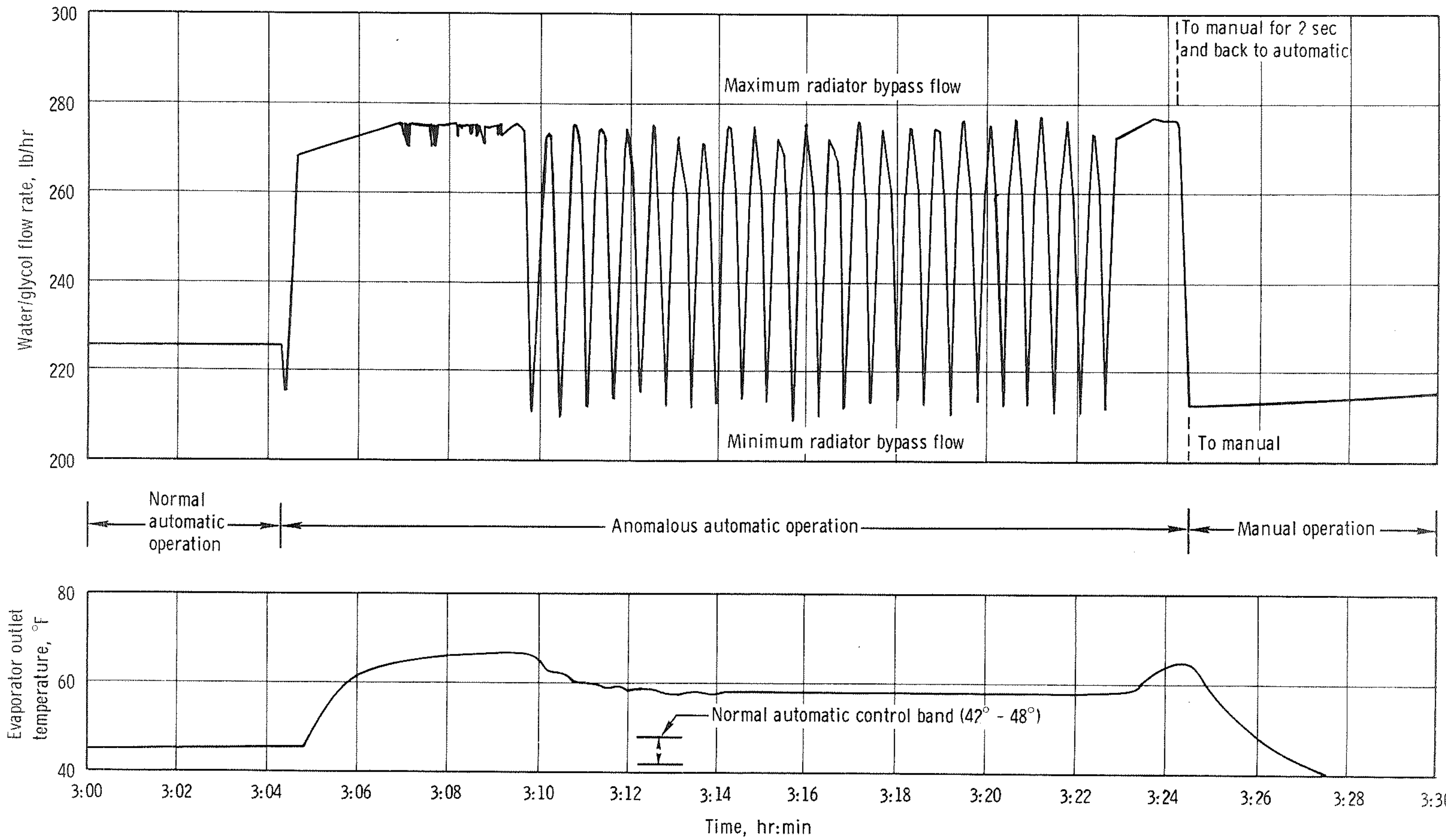


Figure 2-2. - Evaporator outlet temperature and system flow rate during anomalous period.

Previous experience with this type of transducer showed that the reference cavity can leak through the header. Failure histories show that leaks can occur through the feed-through tubes in the header and at the header-to-case weld (fig. 2-3). During altitude chamber tests for Apollo 16 and during the Apollo 12 mission, a similar transducer showed the same type of failure. The Apollo 16 transducer (replaced after the altitude chamber tests) had not had a vacuum check since 1965.

The corrective action will be to require a vacuum check of all installed or replacement pressure transducers which have a sealed reference cavity.

This anomaly is closed.

2.3 ERRONEOUS GIMBAL LOCK INDICATION

At 38:18:56, the computer issued a gimbal lock indication when no gimbal lock condition existed. The vehicle was in the passive thermal control mode, rolling at minus 0.3 degree per second, and the computer was in the platform realignment program at that time. At 38:18:54.7, the computer idling program was selected. This caused the thrust vector enable discrete to be removed from the coupling display unit assembly. Removing the discrete causes a relay to switch the source of an 800-hertz excitation voltage. Interrupting the 800-hertz voltage caused a transient which was electromagnetically coupled into the middle gimbal angle coupling display unit read counter which erroneously set the 90-degree bit. The computer sensed the change in the read counter and determined that the middle gimbal angle was greater than 85 degrees (gimbal lock condition). The computer then downmoded the inertial subsystem to the coarse align mode which caged the platform to the present body reference angles.

Figure 2-4 is a simplified representation of the pertinent circuits in the middle gimbal angle coupling display unit. The purpose of the read counter is to store, in digital form, the platform gimbal angle. If the gimbal angle changes, the analog signal is no longer the same as the digital signal from the read counter and the resulting error signal causes a pulse train to be generated. The pulse train increments or decrements the read counter until it matches the analog gimbal angle. The read counter is a series of flip-flop logic circuits whose only inputs are a pulse train and timing pulses. The 90-degree bit is unique, however, because it can also be set by an ambiguity discrete. The purpose of the ambiguity discrete is to prevent the platform from going to an ambiguous position when the inertial measurement unit is turned on.

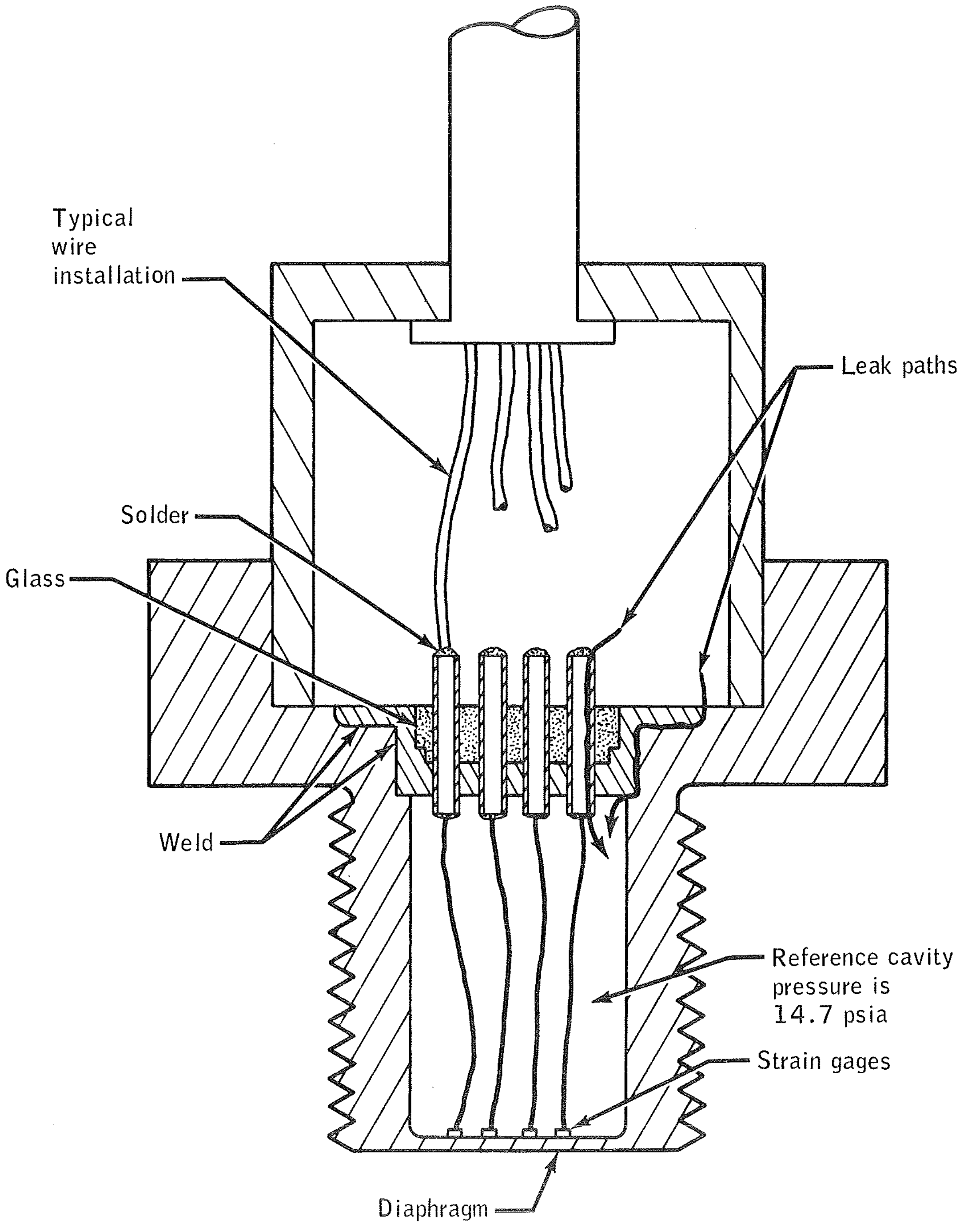


Figure 2-3.- Service propulsion system pressure transducer failure.

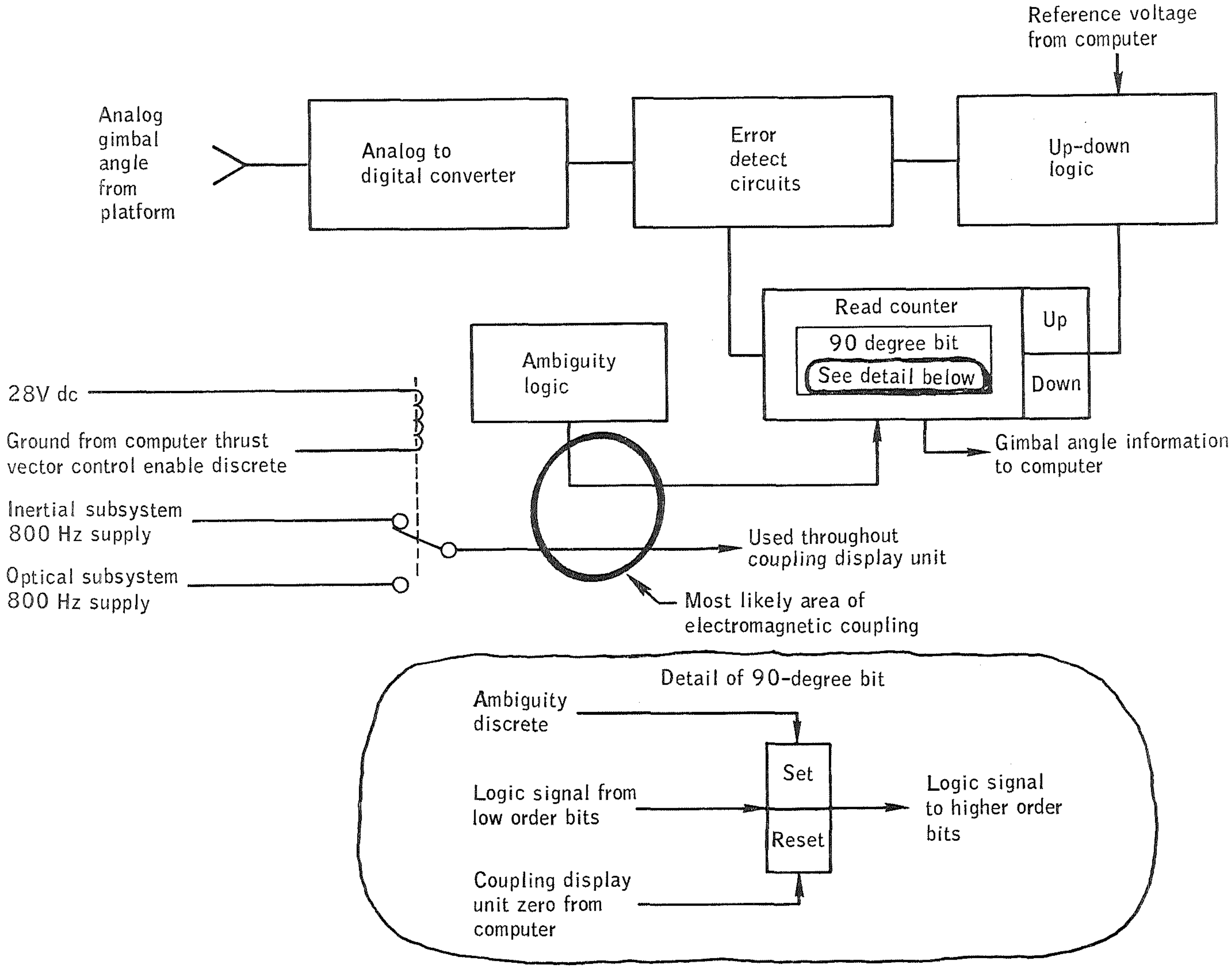


Figure 2-4.- Coupling display unit functional diagram.

The cause of the problem was an erroneous ambiguous discrete which set the 90-degree bit in the middle gimbal angle read counter. The discrete was apparently caused by a transient generated when the 800-hertz excitation voltage was switched from one source to another. A transient caused by switching power sources is highly improbable since it is dependent upon timing of the relay activity in relation to the 800-hertz voltage.

The erroneous gimbal lock problem was circumvented in flight by the use of an erasable program which inhibited the computer from changing the status of the coarse align relay while the erasable program was loaded in the computer.

No hardware changes are anticipated.

This anomaly is open.

2.4 INERTIAL SUBSYSTEM WARNINGS AND COUPLING DISPLAY UNIT FAIL INDICATIONS

The inertial subsystem warning light was observed to illuminate six times during the transearth portion of the flight. Each illumination was accompanied by a program warning of an inertial coupling display unit failure. The warning indications were intermittent; the first four cleared themselves, and the last two were removed by bumping an access panel in the lower equipment bay.

The coupling display unit assembly contains failure detection circuits which monitor the performance of each unit and a failure discrete is sent to the computer if one or more of the following conditions exist for more than 3 to 7 seconds. The first three conditions would be directly observable from flight data; the last two may not necessarily be detected from flight data.

a. Coarse error detect - A disagreement of approximately 30-degrees between a read counter and the 1X gimbal resolver.

b. Fine error detect - A disagreement of more than 0.7 degree between a read counter and the 16X gimbal resolver.

c. Cosine $(\theta - \psi)$ - Voltage less than 4 Vrms. This voltage is normally 4 Vrms when the read counter agrees with the resolver voltage.

d. Read counter limit cycle - The read counter changes the direction in which it is counting at a rate greater than 160 times per second.

e. 14-volt-dc power supply - The 14-volt-dc power supply in the coupling display unit decreases to 8 volts dc or less.

The time delay associated with the failure discrete to the computer was measured in flight as approximately 5 seconds. The failure indications persisted for 4, 12, 16, 894, 10, and 38 seconds, respectively.

Data during each occurrence has been reviewed and no abnormalities were observed. If a real failure of the coupling display unit had occurred and lasted as long as 894 seconds, then some data indication should have been observed. For this reason, an intermittent short in the wiring harness or associated connectors shown in figure 2-5 is suspected. However, other possible causes may be an intermittent in the failure detection logic circuits or the input logic to the computer.

Postflight tests on the individual subassemblies are scheduled.

This anomaly is open.

2.5 HIGH HEAT LEAK INTO HYDROGEN TANK 3

The heat leak into cryogenic hydrogen tank 3 was abnormal during the early hours of the mission. Analysis of the Kennedy Space Center data has shown that the heat leak of this tank was normal prior to lift-off. As shown in figure 2-6, the leak decreased from a high value of 15 to 20 Btu/hour to a normal 2 to 6 Btu/hour by 70 hours and remained normal thereafter. All other parameters were normal.

The insulation for this tank consists of several layers of gold-coated H-film in a vacuum annulus that is formed between the inner pressure vessel and the outer shell (fig. 2-7). The primary insulation system is, however, the vacuum annulus. Degradation of this insulating vacuum will cause a significant increase in the heat leak into the tank as shown in figure 2-8. The analysis indicates that the high initial heat leak resulted from a degradation of this vacuum by approximately one decade (approximately 10^{-6} to 10^{-5} torr). This degradation could have been caused by either of two occurrences:

a. If thermal insulation layers rubbed against one another during the launch phase vibration, the condition could have caused the release of previously entrapped or absorbed gas molecules into the annulus.

b. An ambient air leak into the annulus could have occurred during launch.

Note: Heavy arrows indicate points which shorted together would cause a fail indication.

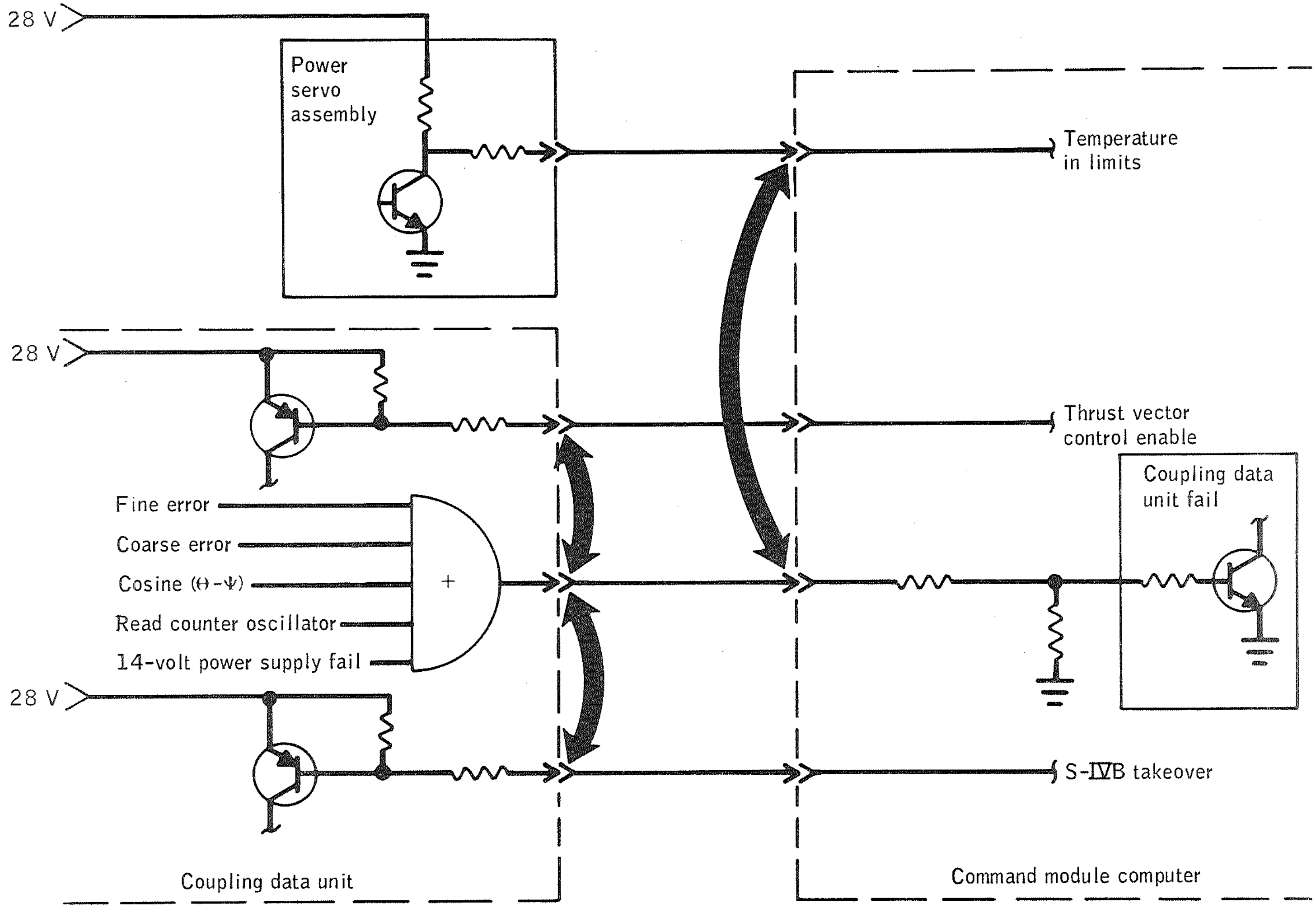


Figure 2-5.- Coupling data unit fail indication.

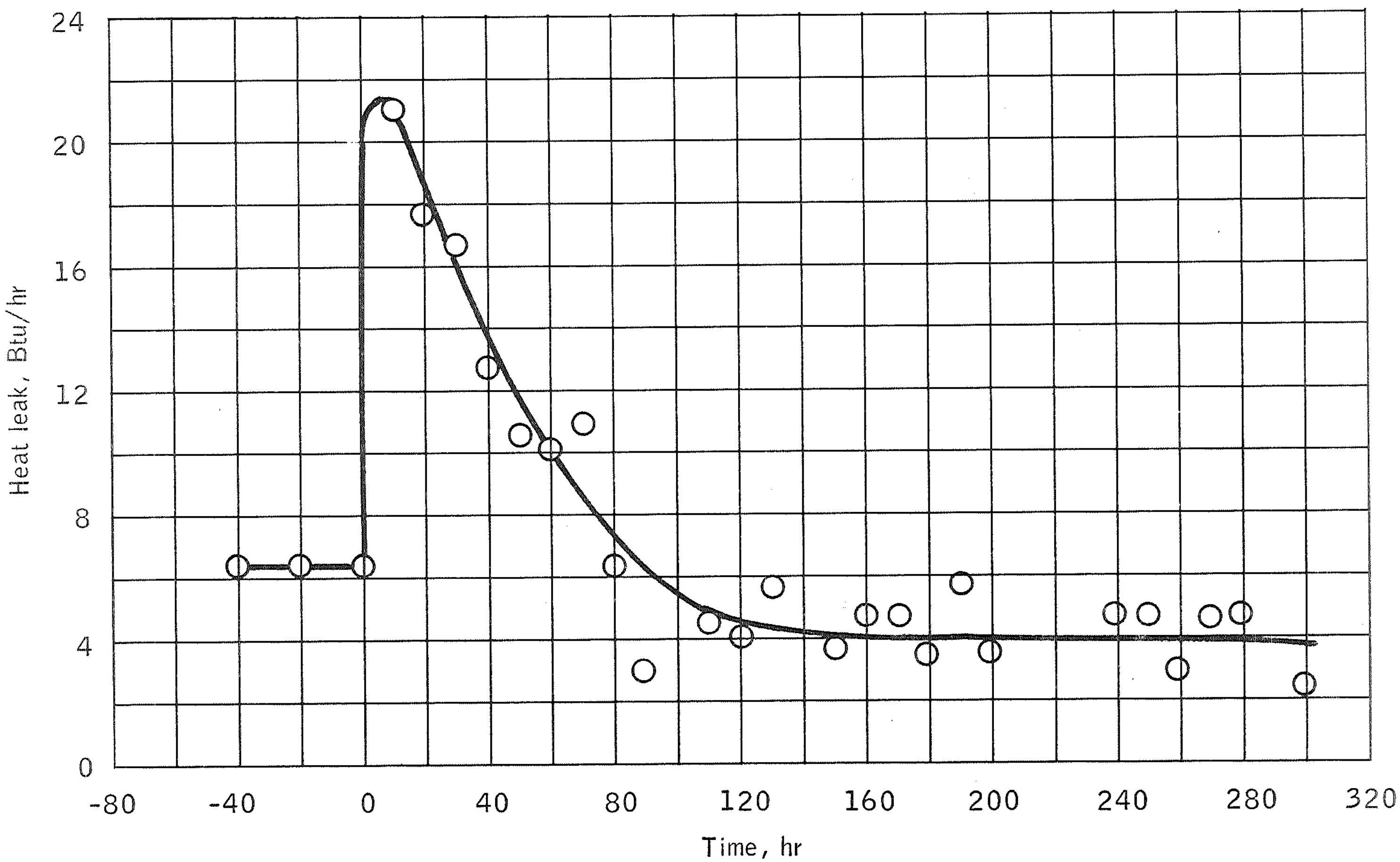


Figure 2-6.- Hydrogen heat leak data.

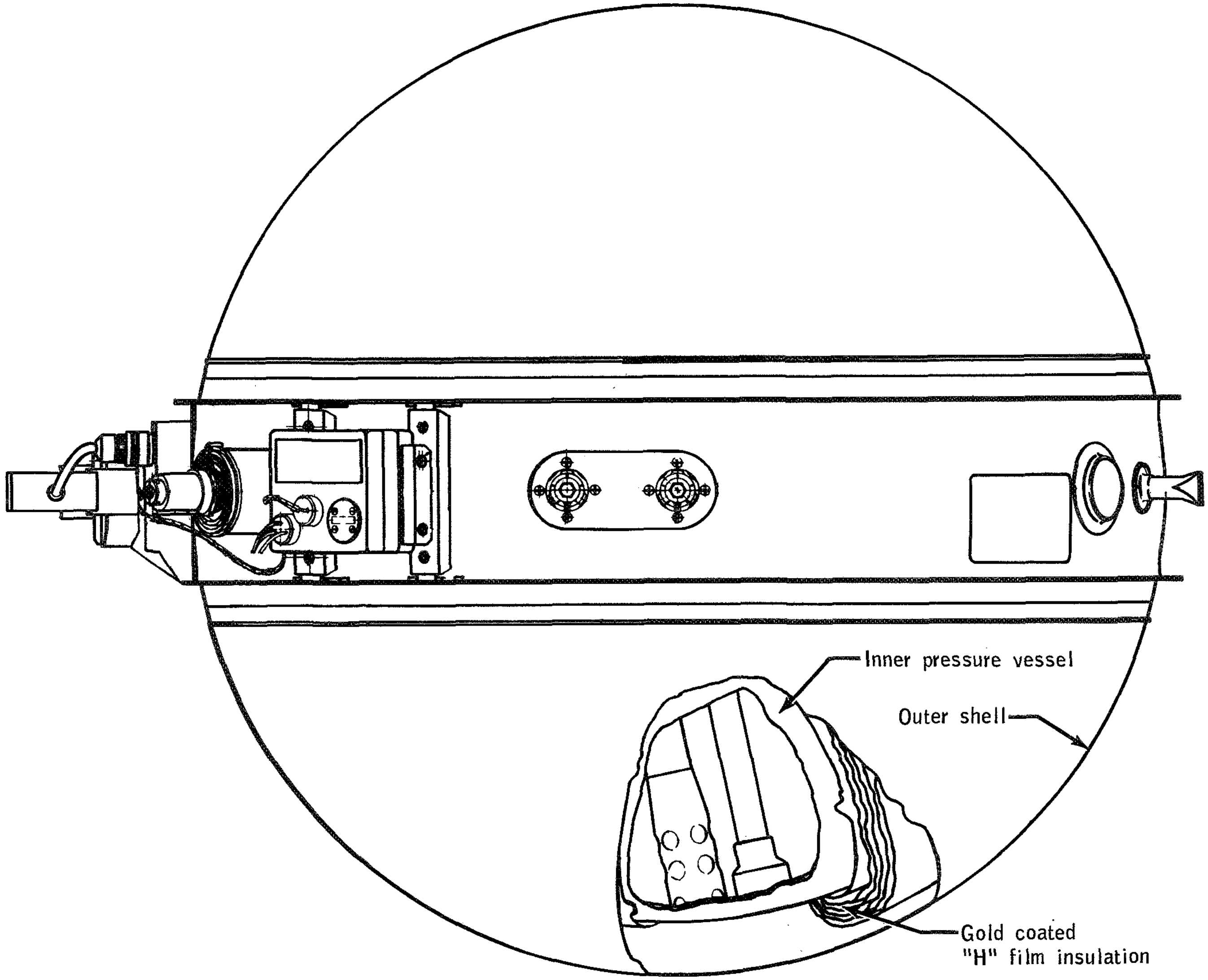


Figure 2-7.- Hydrogen tank.

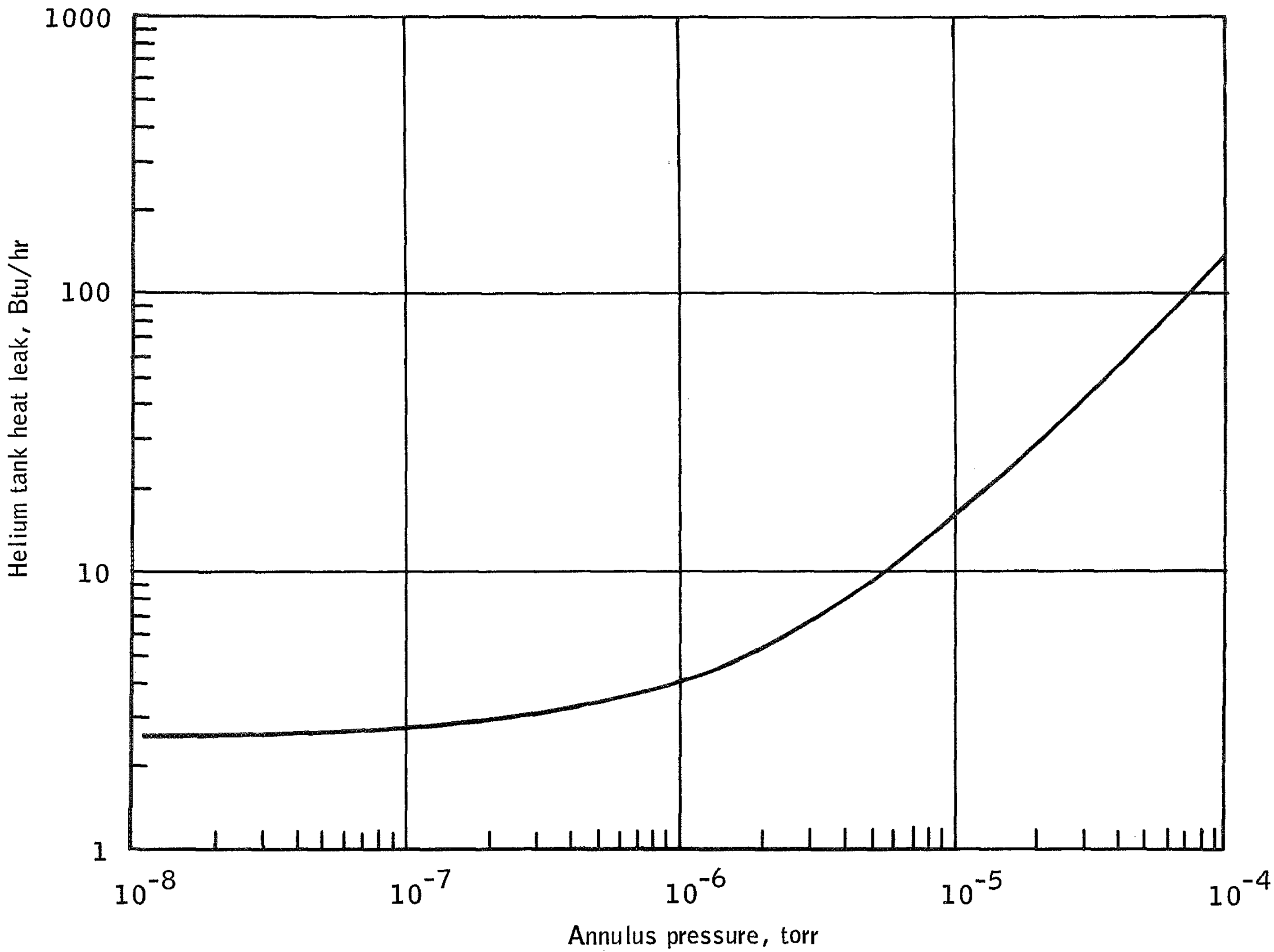


Figure 2-8.- Hydrogen tank heat leak as a function of annulus pressure.

Analysis is continuing to further investigate both of these possibilities. In addition, studies are continuing to verify that pressure oscillations did not contribute to the high heat leak.

A hydrogen leak into the annulus has been ruled out because a hydrogen leak should have continued throughout the mission, and would have caused the heat leak to remain at a high level. The qualification vibration test levels and procedures for the hydrogen tank have been reviewed and are adequate.

This anomaly is open.

2.6 UP-DATA LINK COMMAND CAPABILITY LOSS

The spacecraft failed to respond to uplink real-time commands at 42:23:09 and again at 207:14:36. In each case, the condition existed until the crew cycled the up-telemetry command reset switch. This restored normal operation.

A similar problem was experienced on Apollo 9. Extensive review of the spacecraft data, testing of flight hardware, and analysis of the command system did not identify a specific cause of the problem. A special low-voltage test did show that a low-voltage transient could change the vehicle address register and inhibit receipt of any commands until power is cycled to OFF and back to ON, resetting the register. However, on Apollo 9, telemetry data failed to indicate any voltage transient conditions that could be expected to change the vehicle address.

Tests of the Apollo 16 spacecraft up-data link equipment are planned.

This anomaly is open.

2.7 LEAKAGE DURING INFLIGHT CHLORINATION

During the third daily water system chlorination, difficulty was experienced in injecting the chlorine ampule. (The chlorine ampule is placed in an injector before use.) Advancing (rotating) of the injector knob seemed stiffer than usual and fluid leakage was noted at the interface between the needle assembly and the injector when the injector was removed. This leaking stopped quickly, however, and no loosening of the injection port septum retention insert is suspected. A following buffer injection attempt also appeared to produce some minor leakage. Subsequently, another buffer ampule was injected successfully. Figure 2-9 shows typically

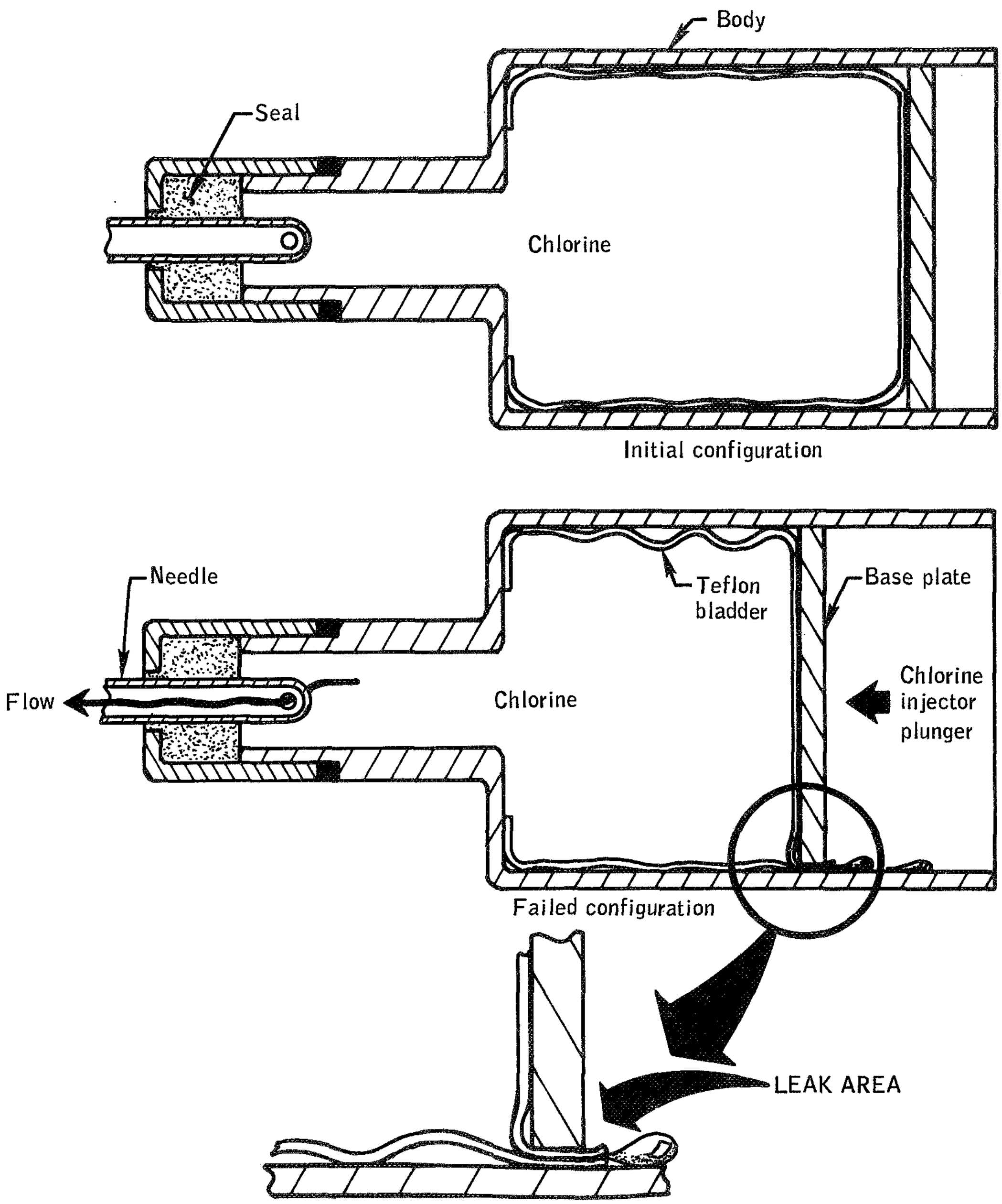


Figure 2-9.- Chlorine ampule failure.

how the Teflon bladder can be forced past the injector base plate. The outer cylinder is not sufficiently rigid to prevent the bladder from extruding around the base plate and being creased, which would be conducive to leaks.

When examined by the crew, the chlorine ampule appeared to be broken (leaking) and the inner bag was visible after having been extruded past the base plate. Inner bag extrusion has been observed before but has not been associated with leaks. The ampule was later jettisoned with the lunar module.

All ampules which were returned with the command module will be inspected for abnormalities and subjected to leak tests.

This anomaly is open.

2.8 FAILURE OF MASS SPECTROMETER BOOM TO FULLY RETRACT

During scientific instrument module experiment operations, the mass spectrometer boom mechanism stalled and would not fully retract. The proximity switches indicated that the mechanism always retracted past the safe service propulsion system firing position except before transearth injection. At that time, the mechanism stalled two-thirds out (approximately 200 inches) and would neither extend nor retract. The boom mechanism and experiment were jettisoned prior to the transearth injection maneuver. The cause of the previous stalls is not known, but bunching or snagging during retraction of the coiled experiment power cable is considered the most probable cause as this had occurred previously during 1-g testing.

The Apollo 17 lunar sounder has two HF antennas which are deployed and retracted by a mechanism that is the same in concept as the mass spectrometer and the gamma ray boom mechanisms (see sec. 2.9). However, the HF antenna mechanism is a simpler design and the antenna has no cables which are required to travel in and out with the antenna.

A review of the HF antenna mechanism design and qualification requirements is being made, and aspects other than cable bunching are being studied.

This anomaly is open.

2.9 GAMMA RAY SPECTROMETER BOOM MECHANISM STALLED

During scientific instrument module experiment operations, the gamma ray spectrometer boom mechanism stalled and would not fully retract on three of five retractions. The boom mechanism always retracted past the safe service propulsion system firing position as indicated by the proximity switch.

The cause of the retract failure is not known, but bunching of the coiled experiment power cable at the annulus of the stowage housing during retraction is considered the most probable reason since this has happened on other occasions during 1-g testing.

Aspects other than cable bunching are being studied.

This anomaly is open.

2.10 SERVICE PROPULSION ENGINE GIMBAL ACTUATOR OSCILLATIONS

During the pre-ignition checklist for the lunar orbit circularization maneuver, an oscillation was detected in the yaw axis of the secondary servo system for the service propulsion engine gimbal actuator. Figure 2-10 is a functional diagram of the servo loop. The oscillation was present in all modes of operation, but only when the secondary yaw servo loop was used. The oscillation was limited at ±1.0 degree and at 2.4 hertz with the gimbal correctly following commands to different positions. Figure 2-11 shows one occasion that the oscillation did not occur when repositioning the engine which implies that the failure is electrical in nature.

Inflight results were matched by simulations of system performance with an open in the rate feedback loop of the yaw servo system. It was also demonstrated that the oscillations would have damped after ignition because of the side load forces exerted on the engine bell while thrusting. Therefore, the secondary yaw servo system was considered safe to use if the primary servo system were to fail and it became necessary to switch over to the secondary system.

Cause of the failure is an open in the rate feedback loop of the secondary yaw servo system. The open could have occurred in the rate transducer, spacecraft wiring and associated connectors, or in the thrust vector servo assembly electronics package. Postflight testing of the command module portion of the wiring and of the electronics package will be performed in an attempt to isolate the open.

This anomaly is open.

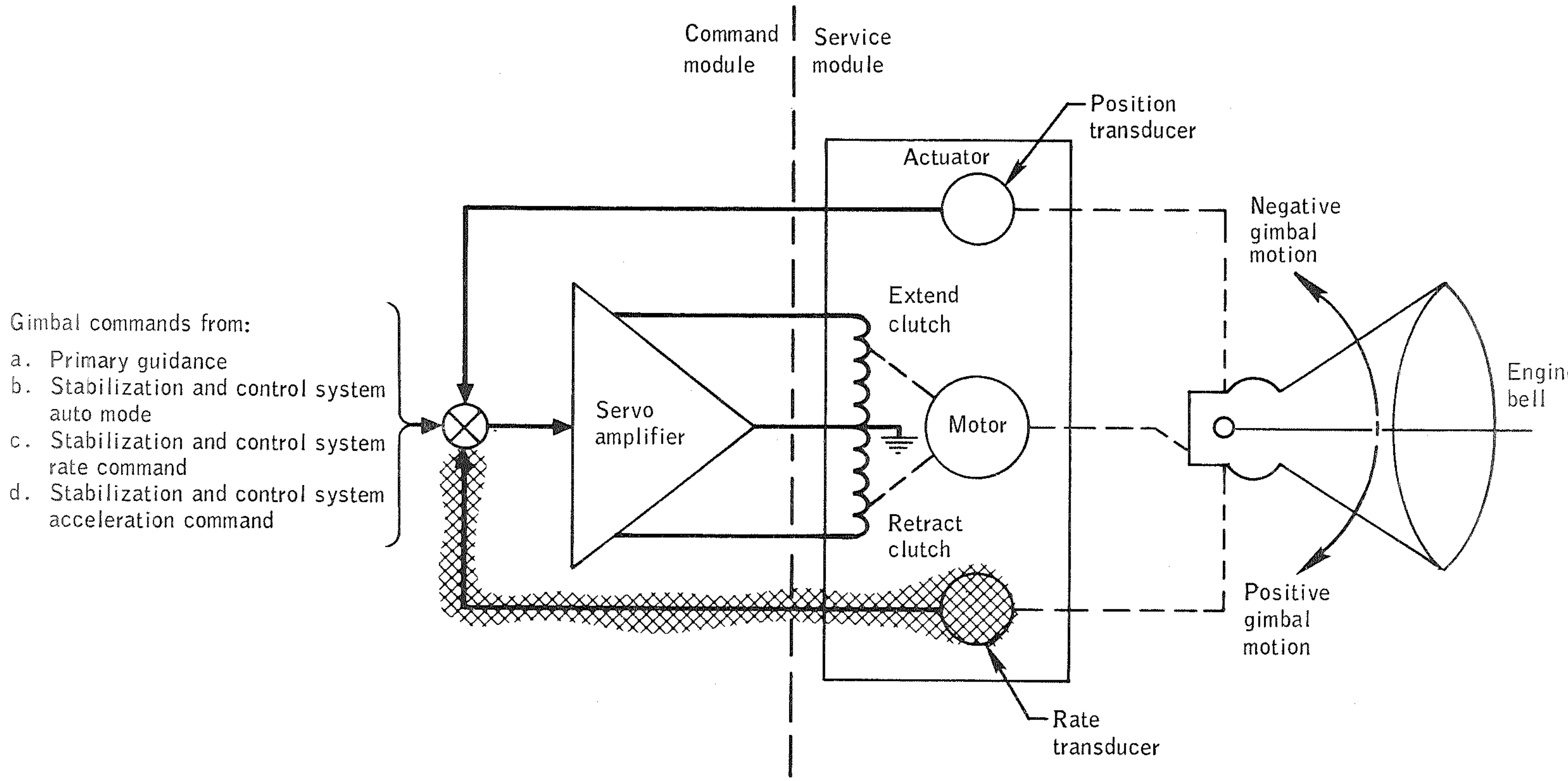


Figure 2-10.- Service propulsion system gimbal servo diagram.

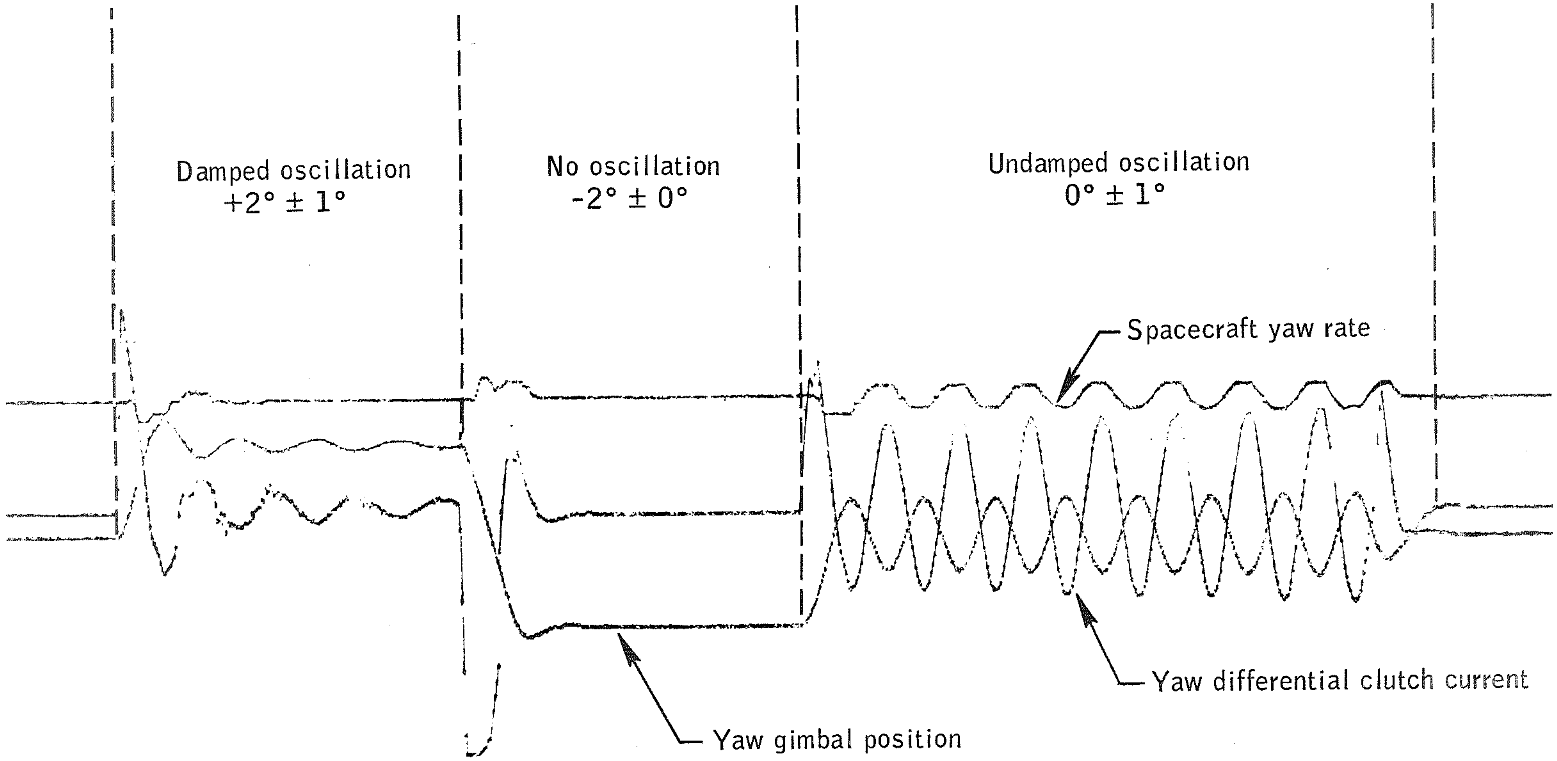


Figure 2-11.- Secondary yaw gimbal drive test (primary guidance control).

2.11 NOISE FROM CABIN FANS

Approximately 24 hours after docking with the lunar module, the command module cabin fans produced a loud moaning sound. The fans were deactivated and not used again during the mission. No cause for the malfunction could be determined by the crew. A filter had been installed for the first time for this mission to provide protection from floating objects entering the outlet of the fan assembly during non-operating periods.

Postflight examination of the fans is scheduled.

This anomaly is open.

2.12 VACUUM CLEANER FAILURE

The crew reported that, after a period of successful operation, the vacuum cleaner failed. The inoperative unit emitted a low hum and was subsequently disconnected and stowed for entry.

Postflight operational tests and analysis are scheduled.

This anomaly is open.

2.13 ERRONEOUS SUIT PRESSURE TRANSDUCER READING

While the cabin was depressurized during the transearth extravehicular activity, the indicated environmental control system suit loop pressure transducer reading was 4.6 psia (fig. 2-12). Specification values for this regulated pressure are 3.5 to 4.0 psia and the control point determined from Kennedy Space Center altitude chamber testing was 3.8 psia. The Commander's and Lunar Module Pilot's suit cuff gage readings of 3.55 and 3.95 psia (sec. 4.6) indicated that pressure regulation was satisfactory and that the transducer reading was erroneous.

Prior to the reduced cabin pressure operations, the suit transducer performed acceptably and read within 0.1 psi of the cabin pressure transducer with the suit loop open to cabin. After cabin repressurization, the suit transducer again appeared to read correctly.

Postflight tests of the transducer are scheduled.

This anomaly is open.

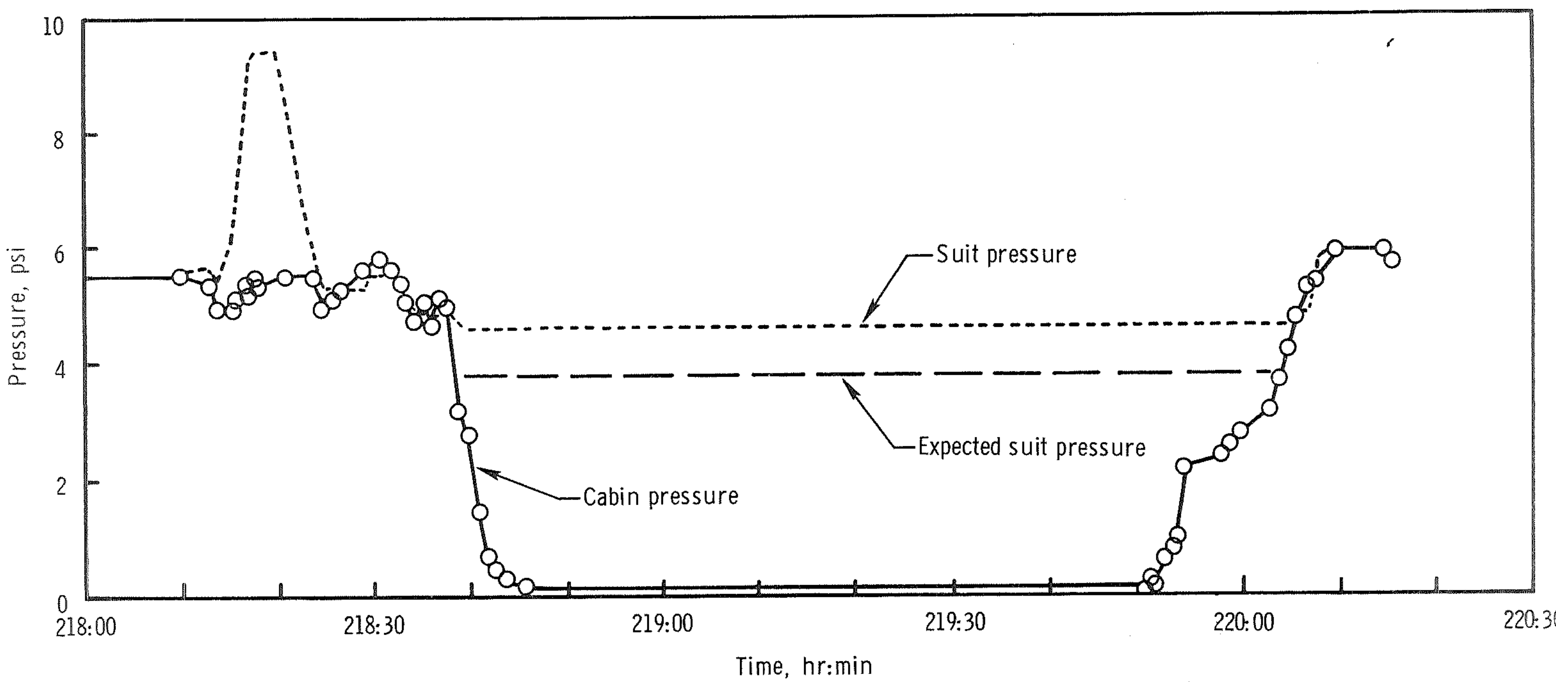


Figure 2-12. - Suit loop pressure data.

2.14 DIGITAL EVENT TIMER COUNTED ERRATICALLY

The digital event timer on panel 1 indicated erroneous time (compared to computer time) after counting up or down over a preset time interval. The digital event timer performed satisfactorily during the early portion of the mission, but began to malfunction approximately midway through the flight.

A similar problem occurred on Apollo 10 and was caused by nonconductive particles becoming momentarily lodged between the brushes and slip ring tabs of the counter wheels (fig. 2-13). These were paint particles from a number counter wheel which was abraided by the idler gear.

Postflight testing of the digital event timer is scheduled.

This anomaly is open.

2.15 UNEVEN DRIVE RATES OF THE SCANNING TELESCOPE

The crew reported that the scanning telescope shaft axis drove erratically and seemed to get worse with time. The condition was observed in the computer mode and the zero optics mode and only through the telescope. The uneven drive rate was related to a change in the characteristic noise level when the optics was being driven. The crew could not be certain whether the uneven drive rates were directly or inversely proportional to the noise level.

The telescope shaft axis is slaved to the sextant shaft through 1X resolvers in each unit. Since no drive problem was reported with the sextant and no uneven drive rates have been observed in flight data, the most likely sources of the problem are the telescope motor tachometer, the motor drive amplifier electronics, the telescope gear train assembly, and the trunnion position angle counter.

Postflight testing of the optical unit is scheduled.

This anomaly is open.

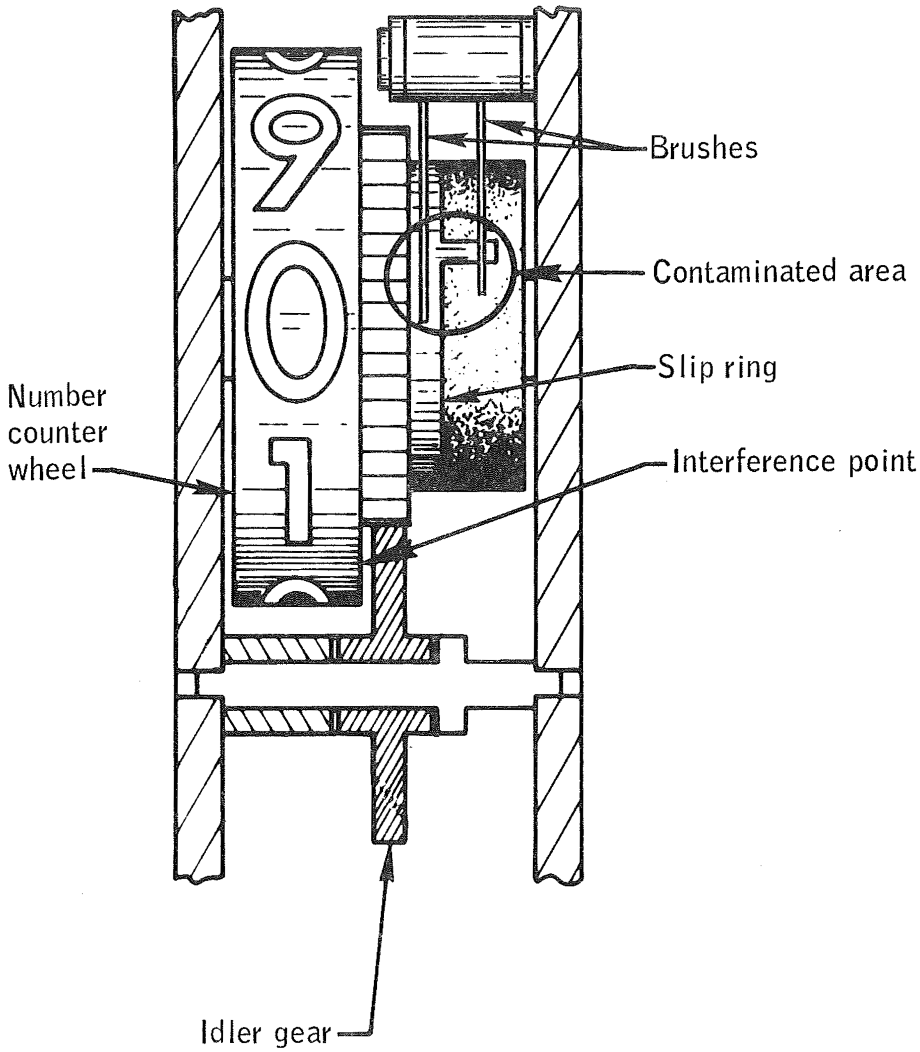


Figure 2-13.- Digital event timer seconds counter.

2.16 STICKING LITHIUM HYDROXIDE ELEMENT

During the first lithium hydroxide element change performed after redocking with the lunar module, considerable difficulty was experienced in removing a used element from side B of the environmental control unit canister. Some minor sticking was also noted during an earlier regular change. No problems were noted with other elements.

Both of the sticking elements had been used extensively during lunar orbit solo operations and probably had expanded due to retention of excess moisture. Under conditions of high flow, low temperature, and low carbon dioxide supply, the exothermic reaction heat is insufficient to drive off the atmospheric and reaction moisture. Such conditions may have existed during this period.

Although the first element was jettisoned with the lunar module, the second element was returned for analysis. Postflight inspection of this element revealed a bulge (fig. 2-14). To accomodate some element swelling, the elements and environmental control unit canister are sized to provide a minimum clearance of 0.030 inch on all four sides of a new element when installed. In addition, all flight elements are fit checked in each canister side with 0.030-inch-thick shims in place prior to launch and removal force is recorded (not to exceed 30 lb). However, the exterior skin of the element has little rigidity and "oil-cans" very easily.

Postflight testing is being performed on the sticking element.

This anomaly is open.

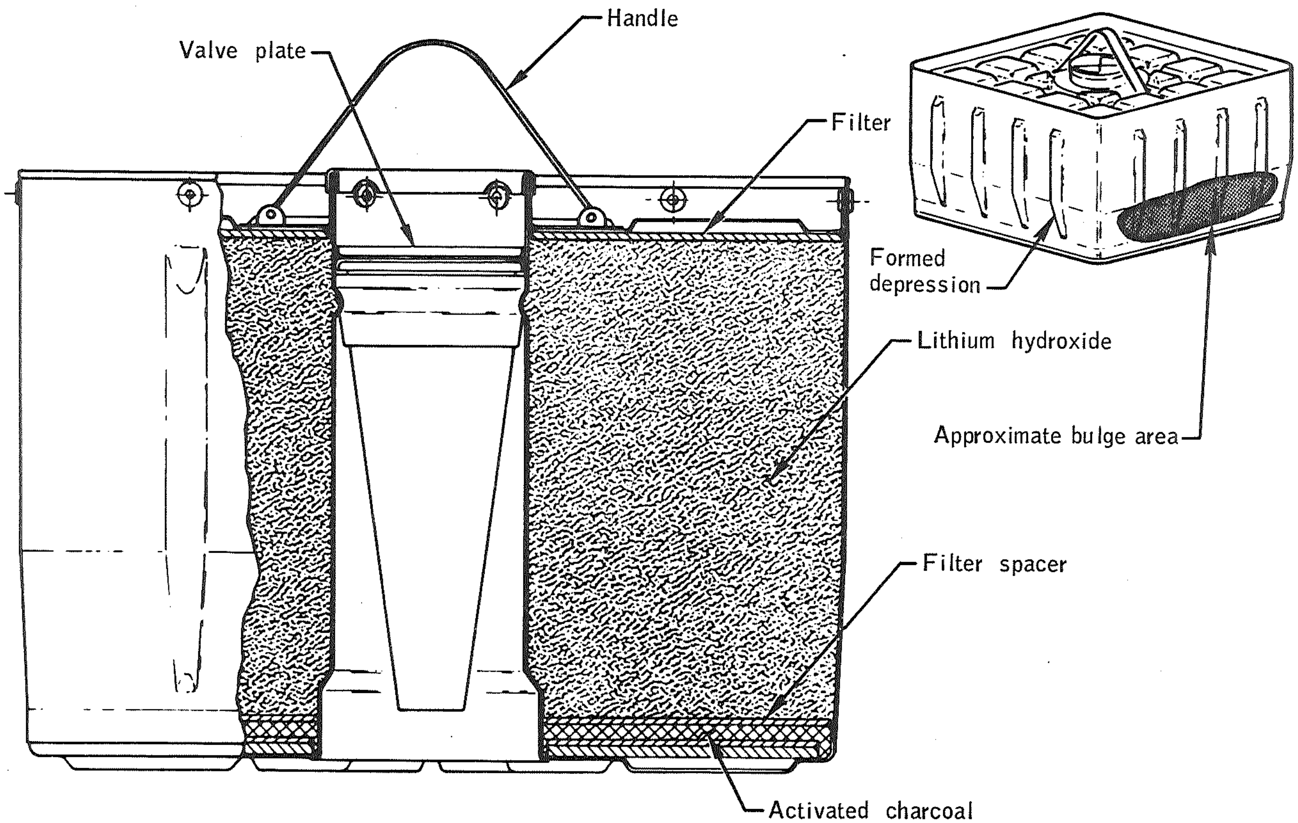


Figure 2-14.- Lithium hydroxide element.

3.0 LUNAR MODULE ANOMALIES

3.1 PAINT FLAKING FROM THERMAL SHIELD PANELS

Just prior to transposition and docking, particles were coming off the thermal shield panels on the minus Y side of the ascent stage. The particles caused the panels to have a grass-like appearance as shown in figure 3-1. (The aft equipment rack thermal shields and the insulation dangling from the bottom of the ascent stage shown in figure 3-1 are discussed in section 3.2.) The paint flakes interferred with star sightings and were potential optical surface contaminants. Thermal tests conducted on specimens removed from a lunar module panel demonstrated that the paint on the panels starts to peel at approximately minus 120° F. The predicted minimum temperature of the panels during the mission is minus 270° F.

Effective with the Apollo 15 spacecraft, five changes were made to the vehicle to minimize the reaction control system propellant temperatures for the 72-hour lunar stay design case. One change was that 16 panels on the ascent stage were painted white. The most effective of the five changes was the addition of tank insulation. The effect of total loss of the paint on the panels results in a maximum reaction control system propellant temperature increase of approximately 2° F.

The corrective action for Apollo 17 will be to remove the paint from the panels.

This anomaly is closed.

3.2 AFT EQUIPMENT RACK PANELS TORN LOOSE

At lunar lift-off, four vertical thermal shields (fig. 3-1) on the aft equipment rack were torn loose from the lower standoffs and remained attached only at the upper standoffs. This occurrence was observed from the lunar-based television.

The most probable cause of the failure was ascent engine exhaust entering the cavity behind these thermal shields. A cross section of the lower edge of the shields is shown in figure 3-2. The thermal shield extending below the support tube allowed a pressure buildup on the closure shield which exceeded its capability. Once the closure shield failed, the exhaust entered the cavity behind the shield, resulting in a pressure buildup exceeding the capability of the vertical thermal shields.

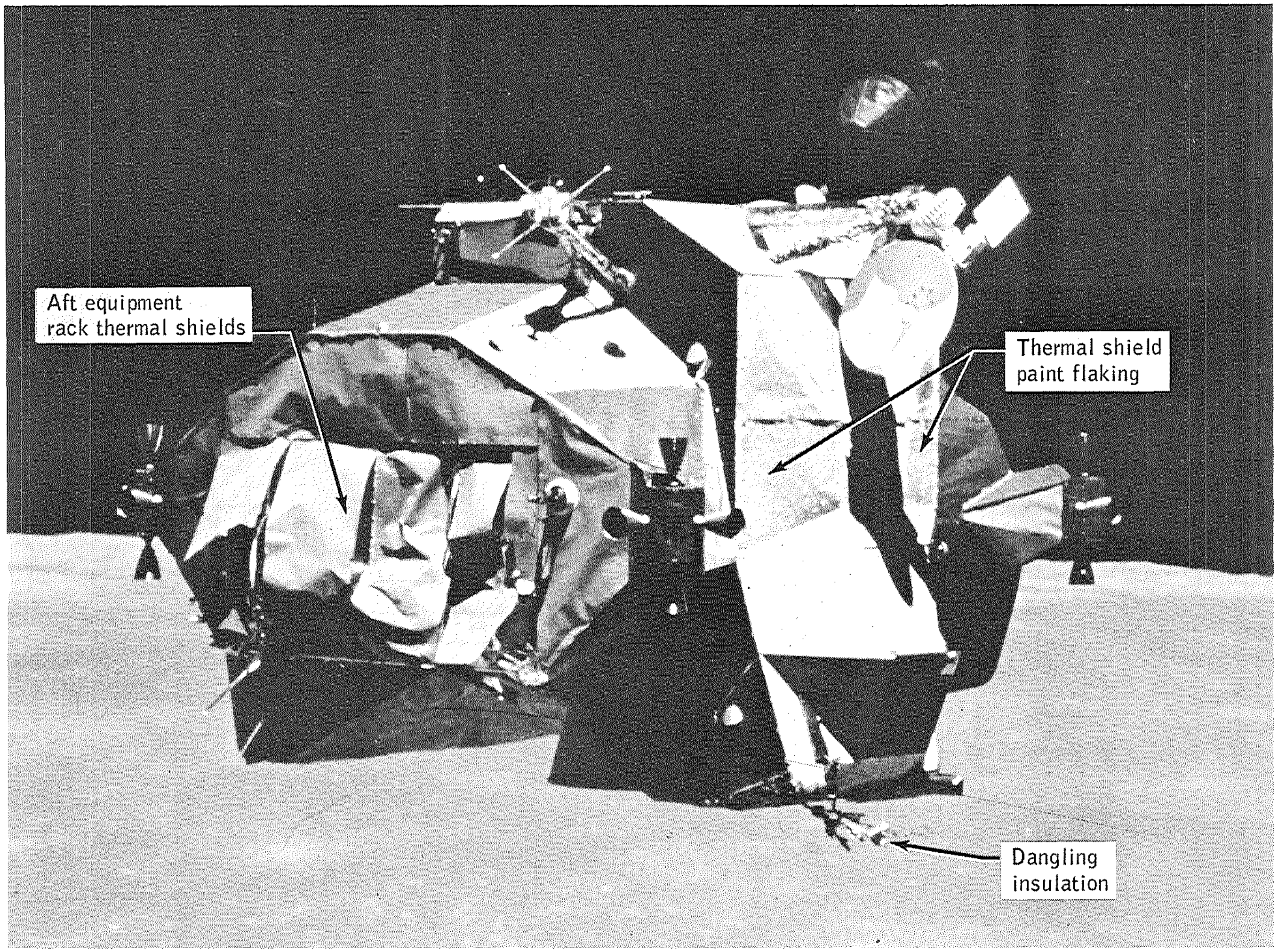
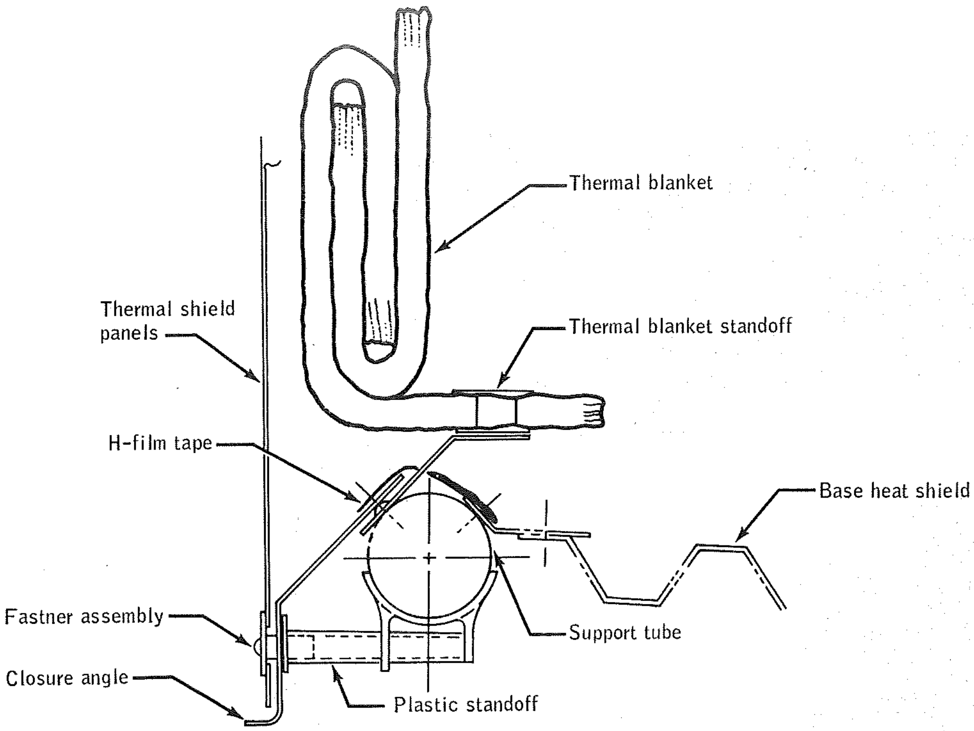
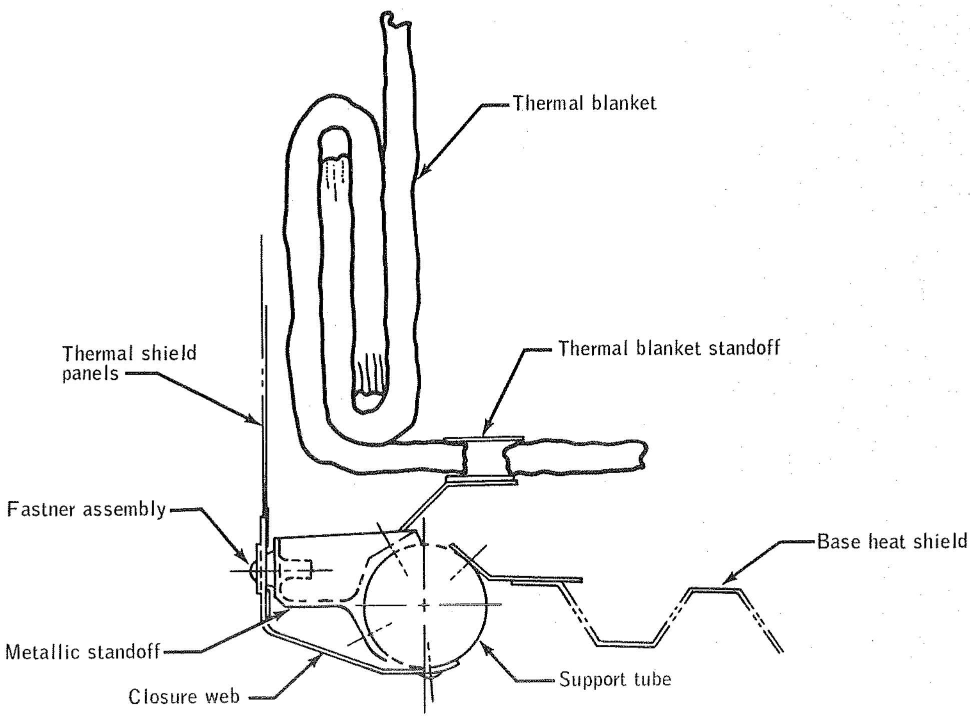


Figure 3-1.- Paint flaking and damaged thermal shield panels



(a) Existing thermal shield configuration.



(b) Proposed thermal shield configuration.

Figure 3-2.- Aft equipment rack thermal protection configuration.

In the lunar surface photographs taken prior to lift-off, some of the shields appear to have come loose from the center standoff (fig. 3-3). Excessive gaps between some of the panels are evident. Both conditions could be caused by excessive pressure in the thermal blanket due to insufficient venting during boost.

The venting configuration of the blanket and adequacy of the vent design criteria are being reviewed. The corrective action planned is to at least redesign the thermal shield to eliminate the projection below the support tube as shown in figure 3-2.

Photographs of the ascent stage taken prior to rendezvous docking (fig. 3-1) show insulation hanging from the plus Y base heat shield. An examination of photographs indicates that the most probable source of the insulation is the thermal shield blanket covering the descent stage blast deflector. This shield is normally blown off during ascent engine ignition; therefore, no corrective action is required for this condition.

This anomaly is open.

3.3 STEERABLE ANTENNA INOPERATIVE IN YAW AXIS

The S-band steerable antenna (fig. 3-4) would not move in the yaw axis during initial activation. Several unsuccessful attempts were made to operate the unit using both the manual slew and auto track modes. The antenna was used in lunar orbit for a short period by changing the vehicle attitude to point the antenna at the earth.

Photographs and inflight test data show that the problem was caused by failure of the mechanical stow latch to release. Functionally, the stow latch restrains the antenna during launch. Figure 3-5 shows the functional operation of the latch. The lock pin, when extended, holds the antenna in the stowed position. A spring is held under compression by a sleeve which is constrained to the latch housing by solder. Melting the solder allows the spring to push the sleeve which, in turn, raises the lock pin out of the retainer. A manual mode is provided for checkout purposes. The lever (fig. 3-5) will pull the lock pin past the ball detent, thus releasing the antenna without disturbing the spring-sleeve-solder configuration.

Antenna photographs, taken on the lunar surface and in lunar orbit prior to docking indicate that the lever for the yaw axis latch was still in the locked position. The crew noted a 2- to 3-degree movement of the yaw indicator during activation attempts. Tolerances in the antenna retainer/locking mechanism allow a slight movement of the antenna dish even though the pin is locked. This condition was duplicated in ground tests, and is normal for a stowed antenna.

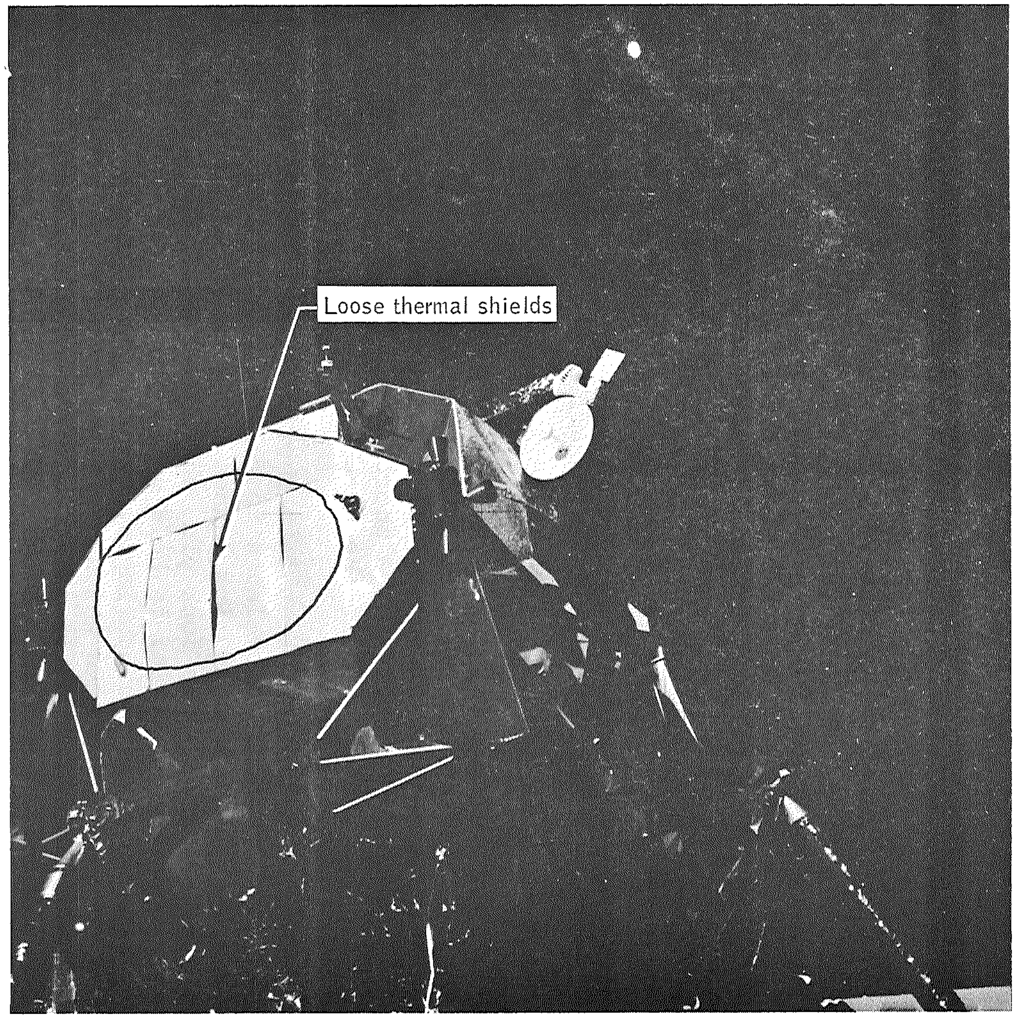


Figure 3-3.- Aft equipment rack thermal shields loose at center standoff.

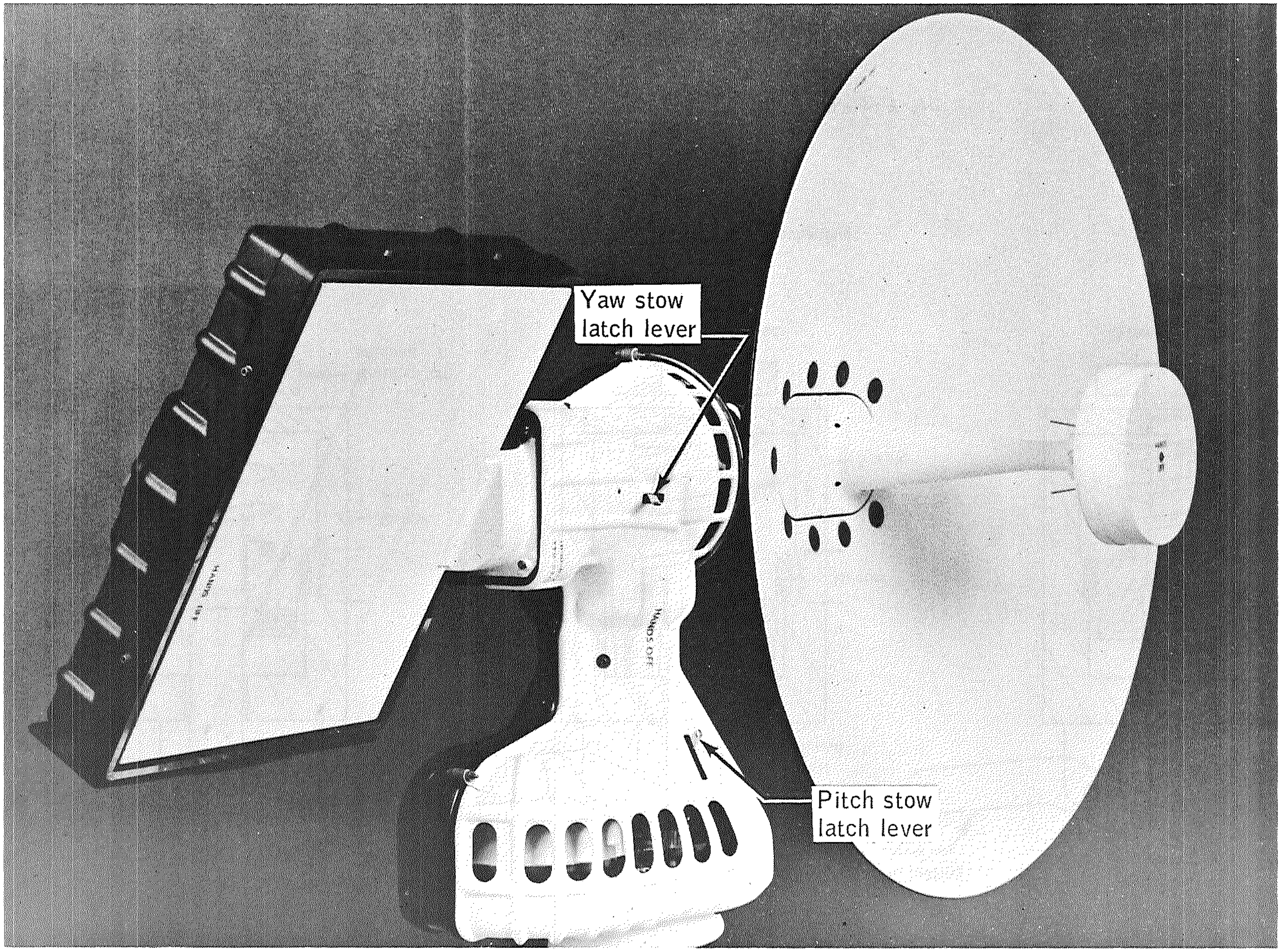


Figure 3-4.- Lunar module S-band steerable antenna.

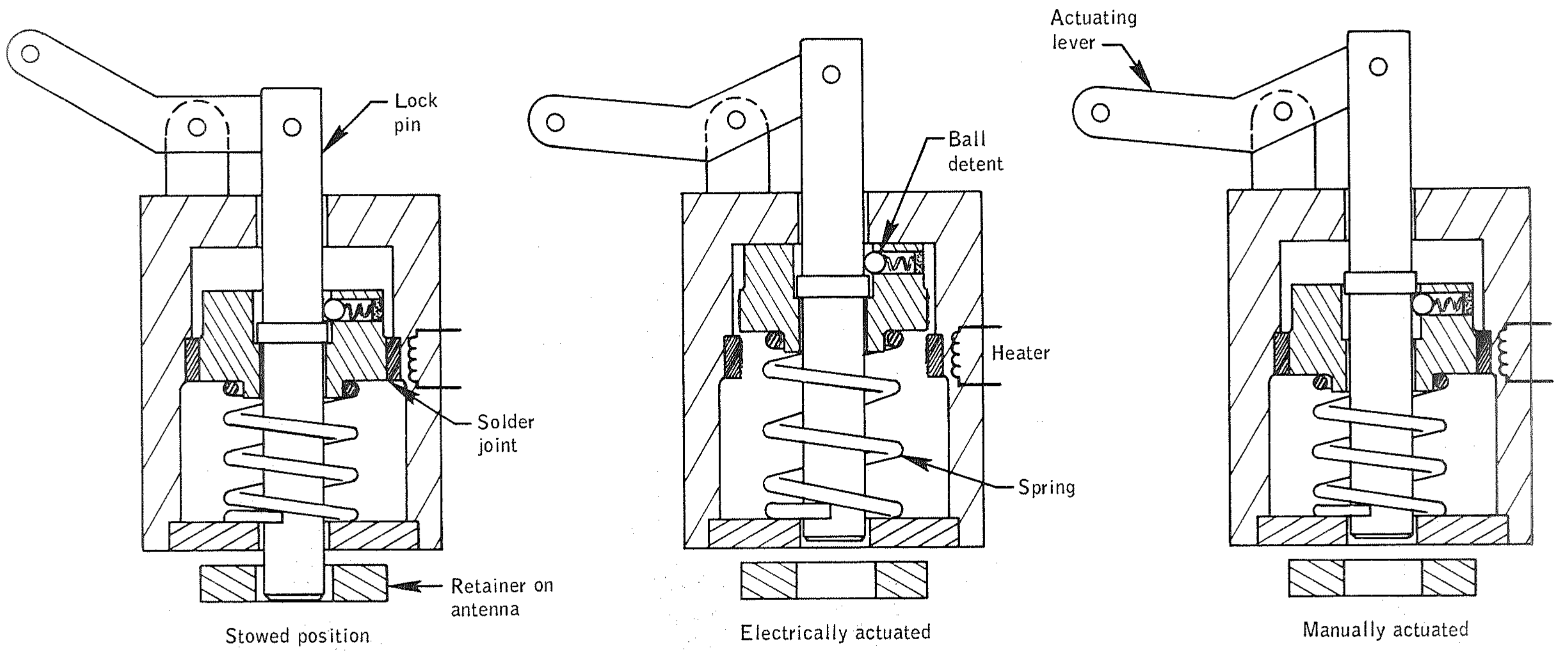


Figure 3-5.- Stow latch mechanism.

There are several areas in the electrical circuitry which could have caused the problem. One is that one contact in the safe/arm switch (figure 3-6) did not make. The second is that an open may have existed downstream of the safe/arm switch, either in the solder melting element or in the wiring back through ground. A resistance measurement is made across the two parallel solder-melting elements prior to flight. About a 3-ohm increase in resistance was found at the last check. This change is equivalent to about a 10-ohm increase in one of the elements which could have been indicative of this problem. Mechanically, the only reasonable failure for this design would be a missing spring or snap ring. The records show no difficulties or problems associated with the entire assembly.

Several corrective actions are planned to cover all reasonable possibilities of failure. These include:

a. X-raying of latch assemblies.

b. Measuring the circuit resistance at Kennedy Space Center.

c. Adding wiring to the circuit to provide redundant paths through the safe/arm switch to the solder melting elements.

This anomaly is open.

3.4 REACTION CONTROL SYSTEM HELIUM REGULATOR LEAKAGE

During pressurization of the system at about 95:00, the regulator outlet pressure in system A (fig. 3-7) increased beyond the maximum specification limit. The normal pressurization is characterized by a sharp increase from the nominal pad pressure, in the interval of 1 to 2 seconds, to the normal regulator outlet pressure. The outlet pressure of system A, after reaching the normal value of 184 psia, continued to rise at approximately 10 psi/minute, a condition which clearly indicated the existence of internal leakage in the regulator assembly. The helium leakage continued throughout the mission, although the leakage rate varied from essentially zero to a maximum of about 1100 scc/min (figs. 3-8 and 3-9). The regulator outlet pressure eventually increased to about 237 psia at which time the relief valves of system A started to periodically relieve the pressure to about 232 psia.

The helium regulator assembly consists of two series regulators (figure 3-10). The unit immediately exposed to the high pressure source is designated as the primary regulator and the downstream unit as the secondary regulator. Both units are identical in configuration with the exception of shims which are used to determine their pressure settings. The primary unit is set to regulate at 181 ±3 psia; the secondary, at 185 ±3 psia.

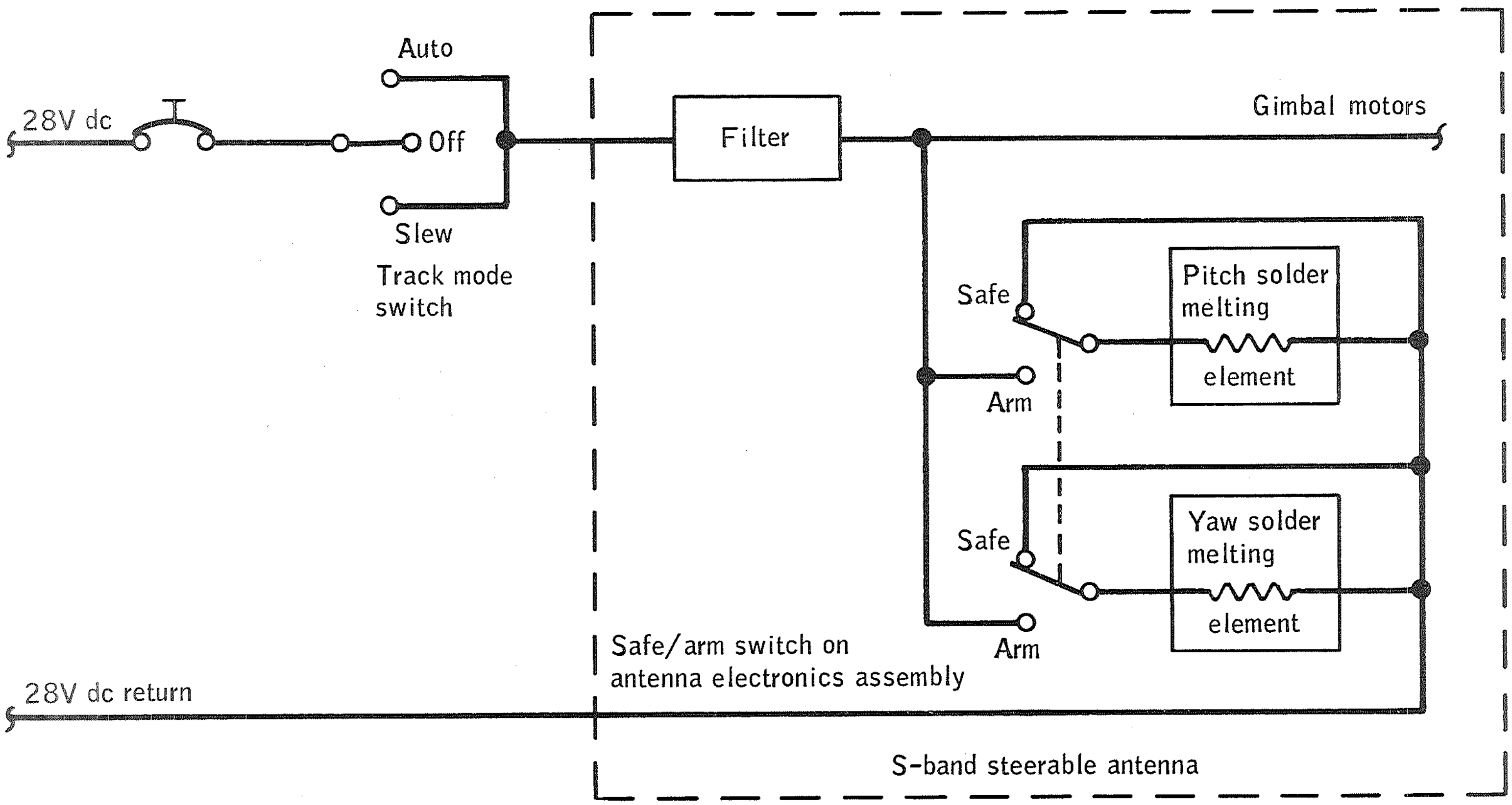


Figure 3-6.- Lunar module steerable antenna stow-latch circuit.

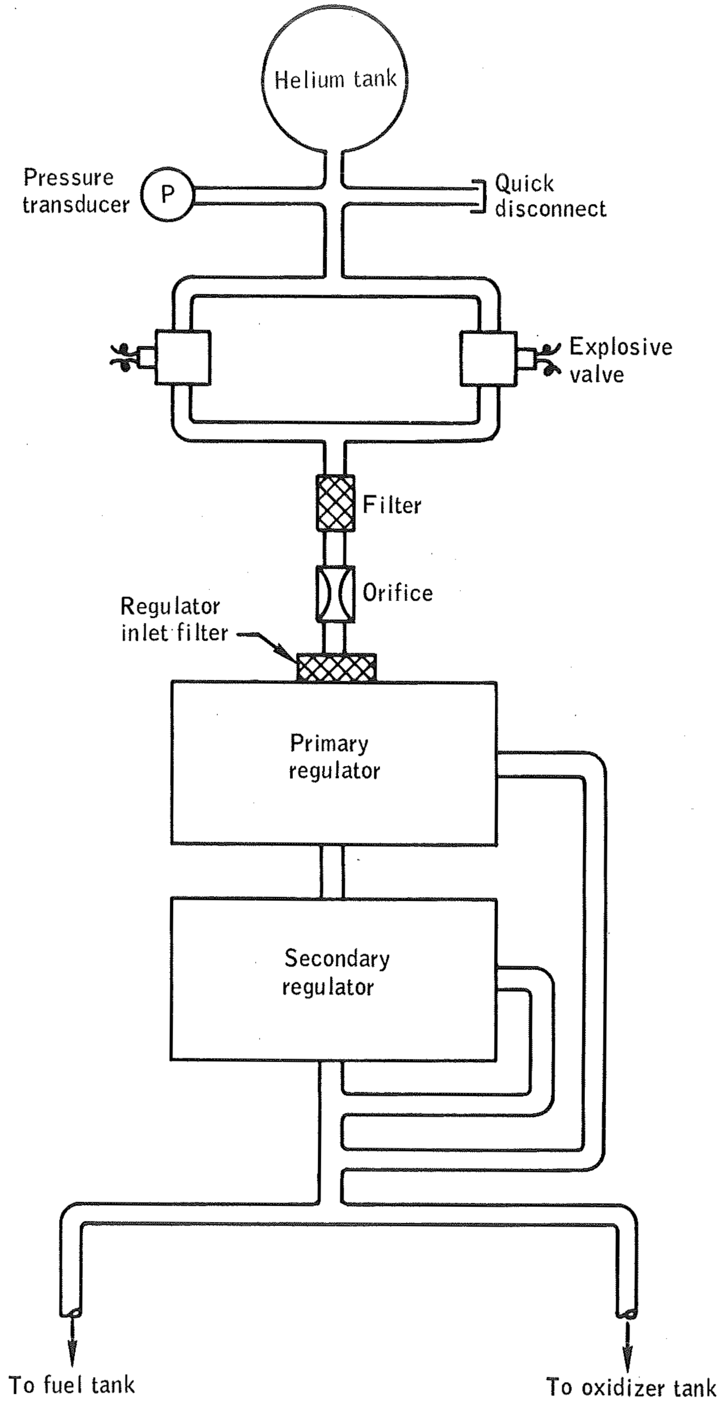


Figure 3-7.- Reaction control system pressurization system.

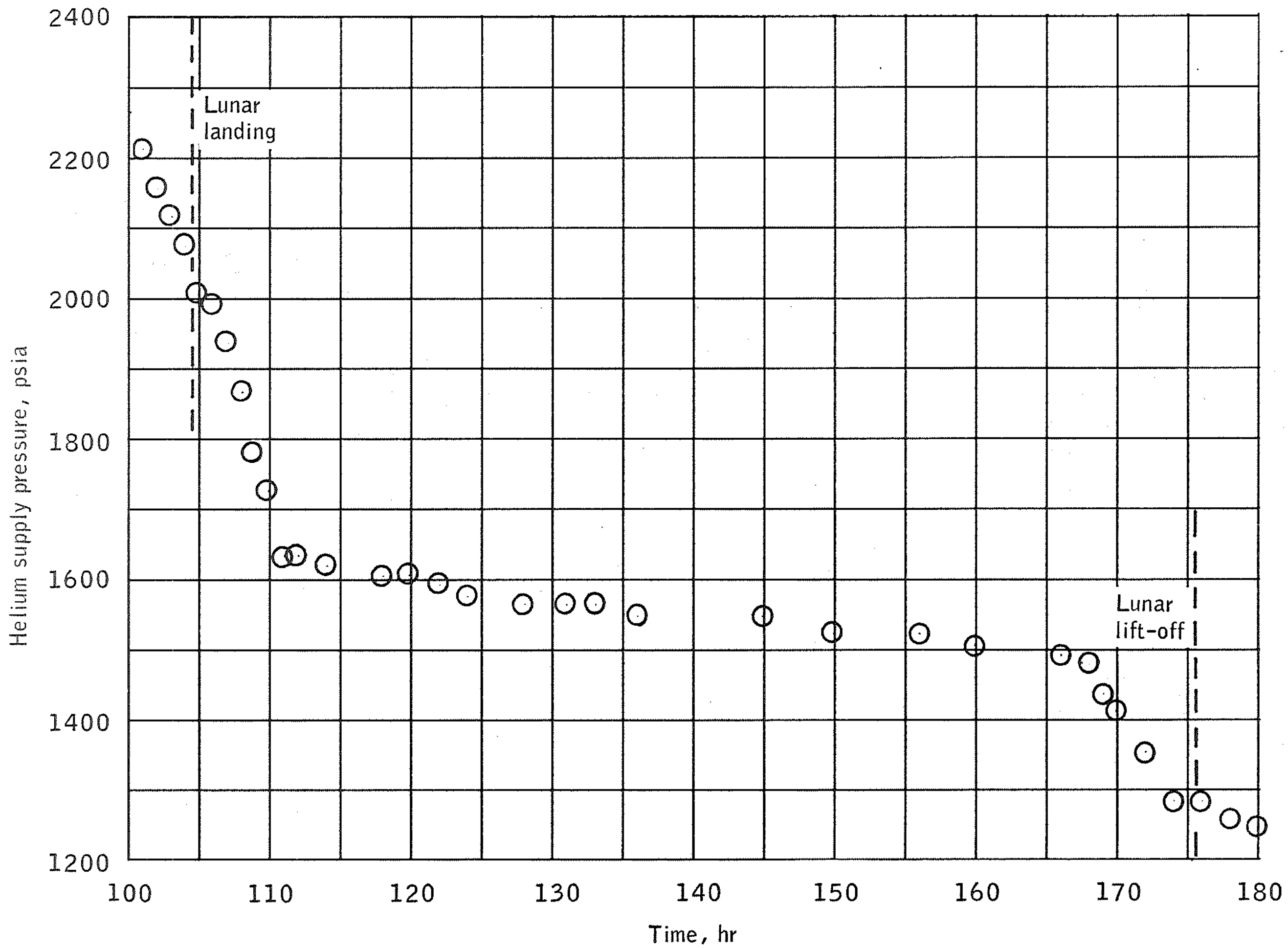


Figure 3-8.- Helium supply pressure for reaction control system during lunar stay.

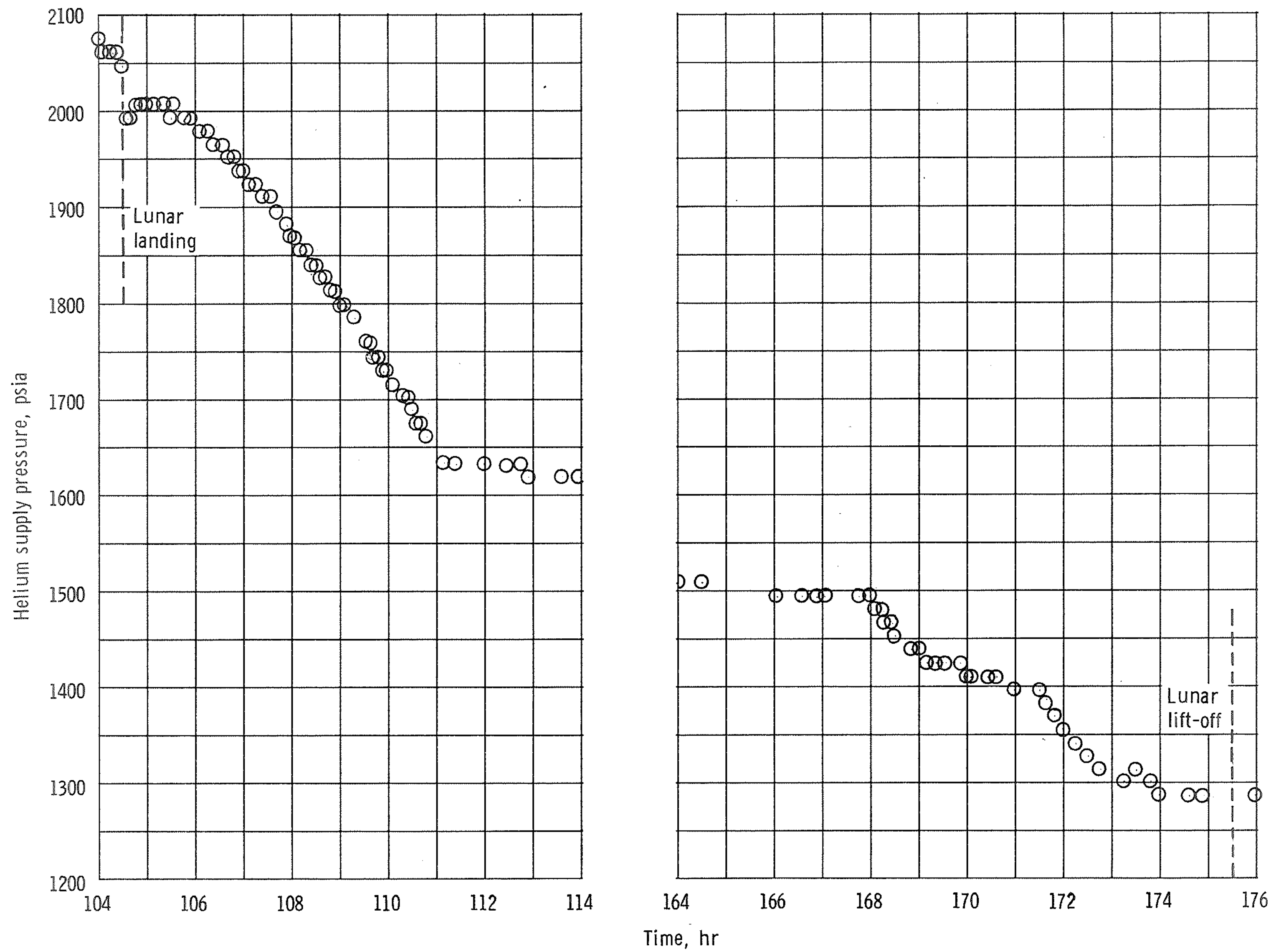


Figure 3-9.- Expanded plot of helium supply pressure during two periods on lunar surface.

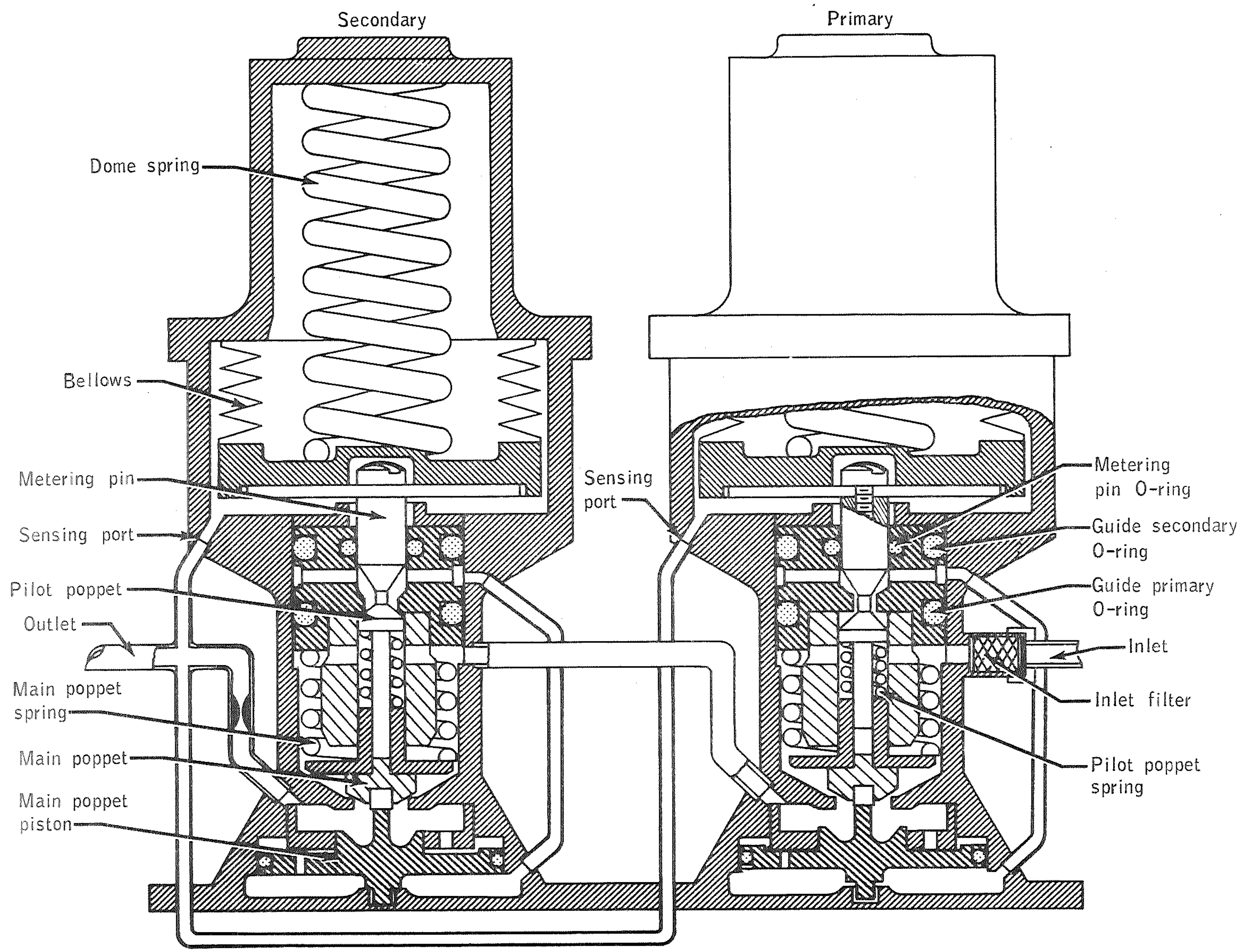


Figure 3-10.- Lunar module reaction control system pressure regulator assembly.

Approximately 3 minutes after pressurization, the first of four propellant transfer operations was performed (fig. 3-11). The propellant transfer operations were performed not only to lower the ullage pressure, but also to create a larger pressurant volume in the propellant tanks, thereby obtaining a blowdown capability for expulsion of the remaining propellant in case of the total loss of system A pressurization gas through the relief valves.

As was mentioned previously, the data indicated that the regulator malfunction was an internal leak beyond specification limits. (Specification value is 20 scc/hr). In order for the leakage through the total assembly to take place, a double failure within the assembly must exist. The matrix of the possible leakage paths is shown in table I. As can be seen from the matrix, there are 15 possible double-failure combinations which can result in the anomalous condition observed during the flight. Of these possible combinations, the four involving only the poppets are deemed the most probable. This is based on the fact that the leakage was variable, indicating a variation in the leakage path. A variable leak path is usually associated with the presence of particulate contamination rather than a damaged O-ring. This supposition is also supported by the fact that all O-rings which could have caused the anomaly were verified to be leak-tight during Kennedy Space Center checkout. The particulate contamination on the other hand could have been inside the regulator assembly during ground tests, but located in such a manner that it was not dislodged by the gas flow under 1-g conditions. In a weightless condition, however, the contamination could have been moved by the gas flow and deposited between any of the poppets and their seats.

It is also possible that, upon system activation, the initial pressure surge through the regulator inlet filter caused filter damage, thereby introducing particulate contamination into the regulator assembly. The flow of the pressurant gas through the regulator during the mission could cause the minute particles to move to and from the poppet seat areas, a condition which would cause a change in the leak paths and the ensuing leakage rates.

Instances of regulator failure that can be traced to particulate contamination are extremely scarce. Of some 1500 units delivered, only a few were found to be leaking because of particulate contamination. All of the contaminated components were discovered during pre-installation tests. The particular component which malfunctioned was delivered during the first quarter of 1966. During pre-installation tests, it was discovered that the secondary outlet pressure was 2 psi below the minimum specification value. The unit was reshimmed, acceptance tested, and pre-installation tested. Subsequently, the unit was brazed into the helium pressurization module and installed on the Apollo 16 lunar module. The regulator was checked two more times prior to shipment to the Kennedy

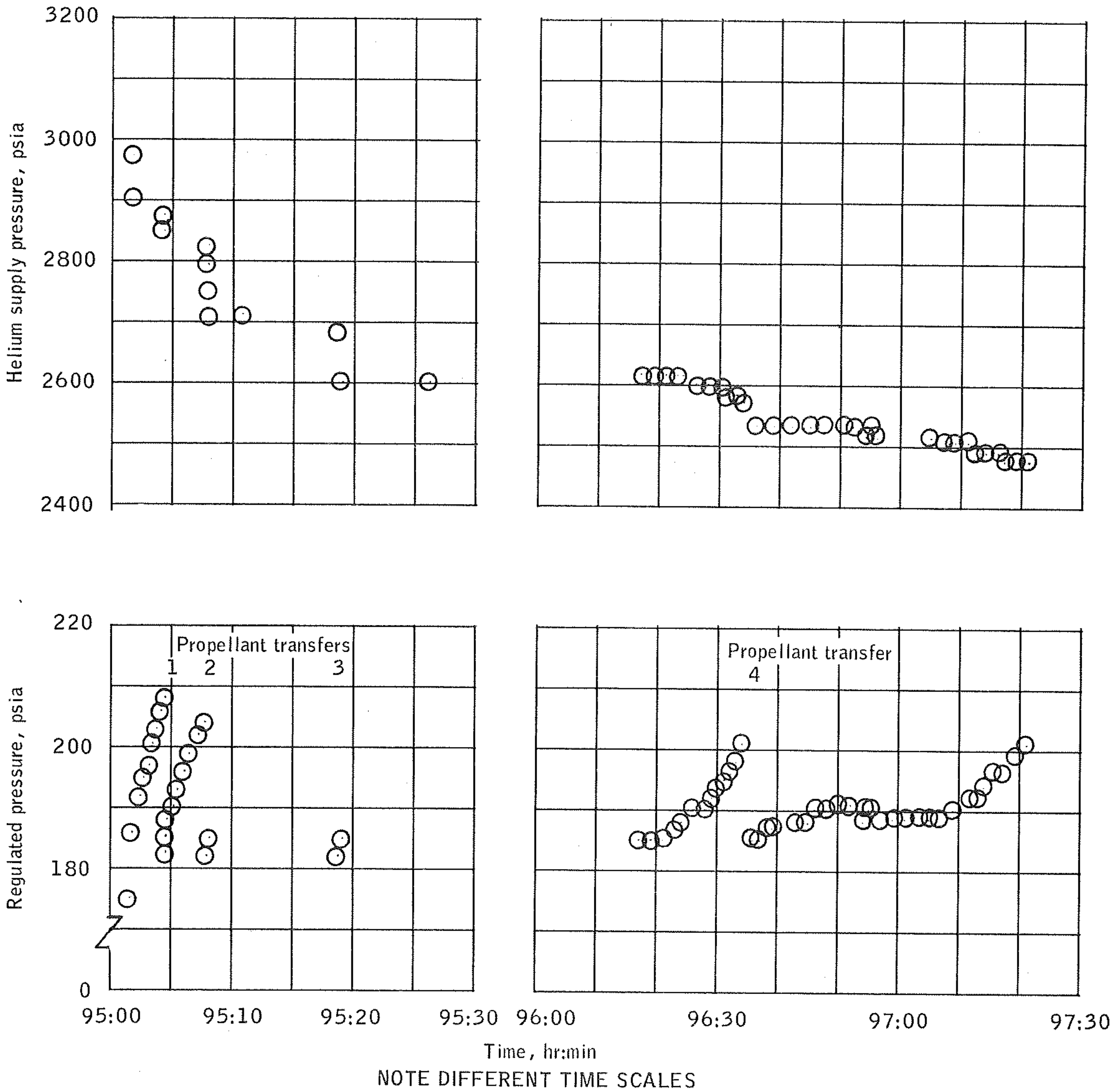


Figure 3-11.- Comparison of helium supply pressure and regulator outlet pressure following reaction control system pressurization.

TABLE I.- MATRIX FOR POSSIBLE HELIUM REGULATOR INTERNAL LEAKAGE

	Secondary regulator pilot poppet	Secondary regulator main poppet	Secondary regulator guide primary O-ring	Primary regulator guide secondary O-ring	Primary regulator metering pin O-ring
Primary regulator pilot poppet	X	X	X	X	X
Primary regulator main poppet	X	X	X	X	X
Primary regulator guide primary O-ring	X	X	X	X	X

Space Center. The initial checkout at the Kennedy Space Center indicated a slight transient overshoot of the lockup pressure of the secondary regulator. This discrepancy was traced to the test setup and the final retest of the regulator indicated that its operation was normal.

A regulator malfunction was detected on module 2 of the S-IVB auxiliary propulsion system. The symptoms associated with the malfunction are not identical to those of the lunar module reaction control system, but the regulators on both vehicles are very similar and, therefore, the S-IVB anomaly may be of interest.

The S-IVB malfunction occurred in two distinct steps. Regulator outlet pressure shifted about 4 psi during the system pressurization some 8 hours before launch. The pressurization of the S-IVB system is accomplished simultaneously with the helium loading and is very gradual. The shift occurred when the regulator inlet pressure was about 400 psia. After that, the regulator outlet pressure remained constant until some 700 seconds after lift-off, when the second step occurred. The pressure began to increase after completion of the first ullage firing. The pressure rise was accompanied by the decrease in the helium source pressure indicating relief valve venting. Calculations indicate that the leakage was a constant orifice flow. The pressure decay of the helium bottle decreased in proportion to the absolute value of the source pressure, thus verifying that a constant orifice flow existed. The leakage persisted throughout the data acquisition period.

The most probable failure mode of the S-IVB regulators is postulated to be particulate contamination. During the pressurization cycle, it could have caused leakage in the primary stage causing the secondary stage to start regulating. During the flight, the contamination could have worked its way into the secondary stage, causing that stage to leak also.

Exploratory tests are underway.

This anomaly is open.

3.5 APPARENT STICKING OF CABIN GAS RETURN VALVE

The crew reported hearing a chattering noise and experiencing a pulsating, insufficient flow in the suit circuit at 95:47 while configured to the cabin mode of operation (fig. 3-12). Data indicate that the cabin mode flow was normal at system activation (93:41), and this was confirmed by the crew after the mission. The pulsing and chattering occurred again after the demand regulator check (95:55) and after lunar landing (105:57).

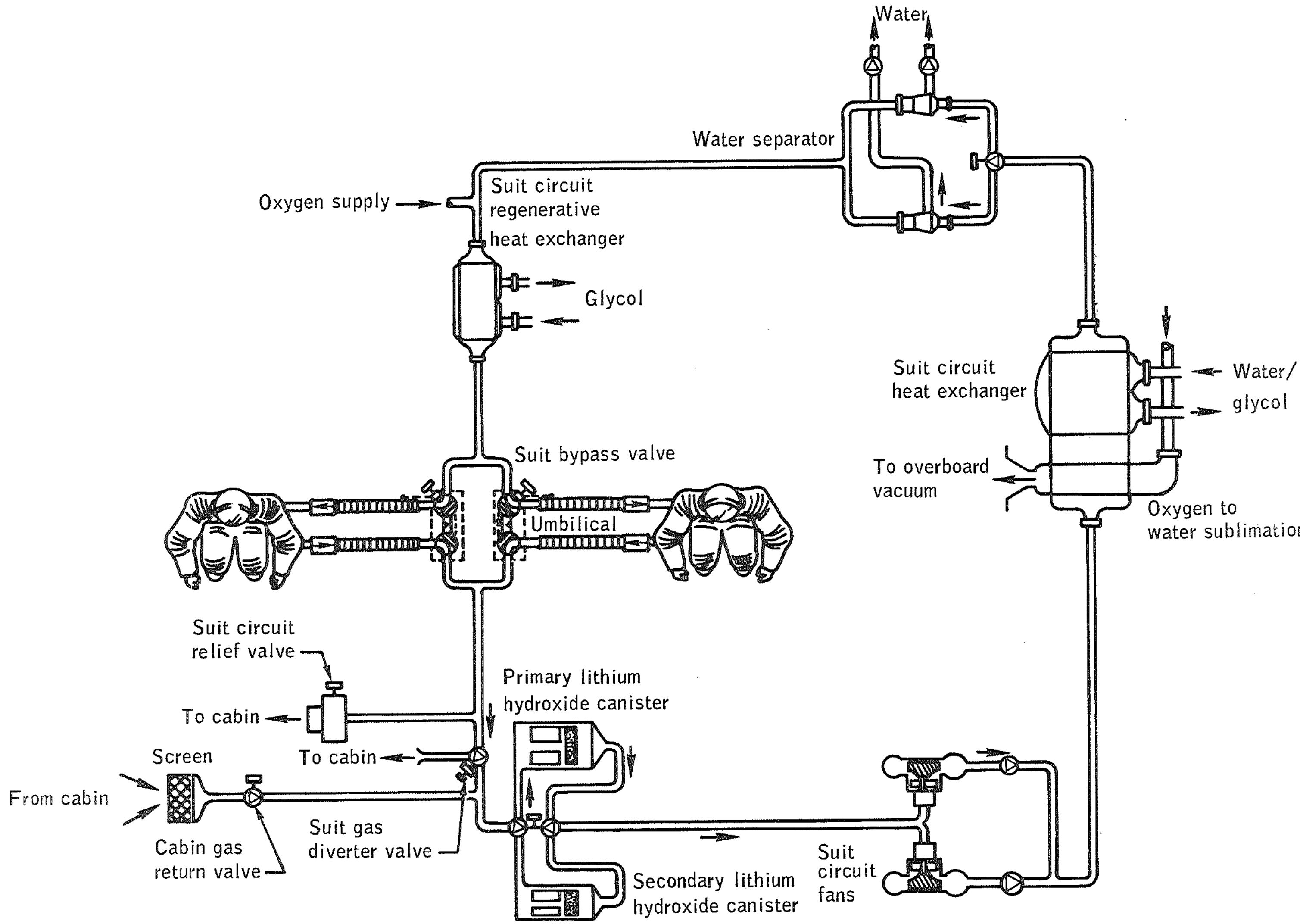


Figure 3-12.- Atmosphere revitalization section.

The problem was traced to the cabin gas return valve which apparently failed to open in the AUTO position, thus blocking the cabin gas from returning to the suit circuit. This results in "deadheading" of the suit fan, thereby causing the downstream check valve to chatter and pulsations due to the small amount of flow drawn through the suit gas diverter valve in the cabin position. The manual OPEN and CLOSED positions of the cabin gas return valve were used for the remainder of the mission. The AUTO position was selected later during the lunar stay and operated normally (good flow, no chatter).

Since the valve functioned properly upon activation and again late in the lunar stay, the apparent cause of the problem was contamination. There are two possible areas of contamination: material on the inlet screen or residue on the flapper seal (fig. 3-13). A suspected source of residue is the orange juice that leaked from the drink bags.

Alternate procedures are available to assure suit circuit flow if a malfunction should occur in the cabin gas return valve. Liquid entering the suit circuit through the cabin gas return valve should not cause a further problem since it would be absorbed in the lithium hydroxide cartridge.

A test will be performed on the Apollo 17 lunar module to check the force needed to open the cabin gas return valve flapper in the AUTO position, thus increasing confidence in the ground checkout of the valve.

This anomaly is closed.

3.6 LOSS OF LUNAR MODULE ATTITUDE CONTROL AFTER JETTISON

Lunar module attitude control was lost immediately after the lunar module was jettisoned from the command and service module at 195:00:12. The lunar module was to be controlled between jettison and lunar surface impact (including the deorbit firing) with the digital autopilot of the primary guidance system (fig. 3-14). After jettison, telemetry showed that the autopilot was properly issuing error signals to the attitude and translation control assembly primary preamplifiers; however, no engine driver commands were being issued by the attitude and translation control assembly, as indicated by telemetry.

Any of the following conditions could have caused the problem (refer to fig. 3-14):

a. The circuit breaker which provides the 28-volt-dc enabling voltage to the primary preamplifiers may have closed mechanically but not electrically.

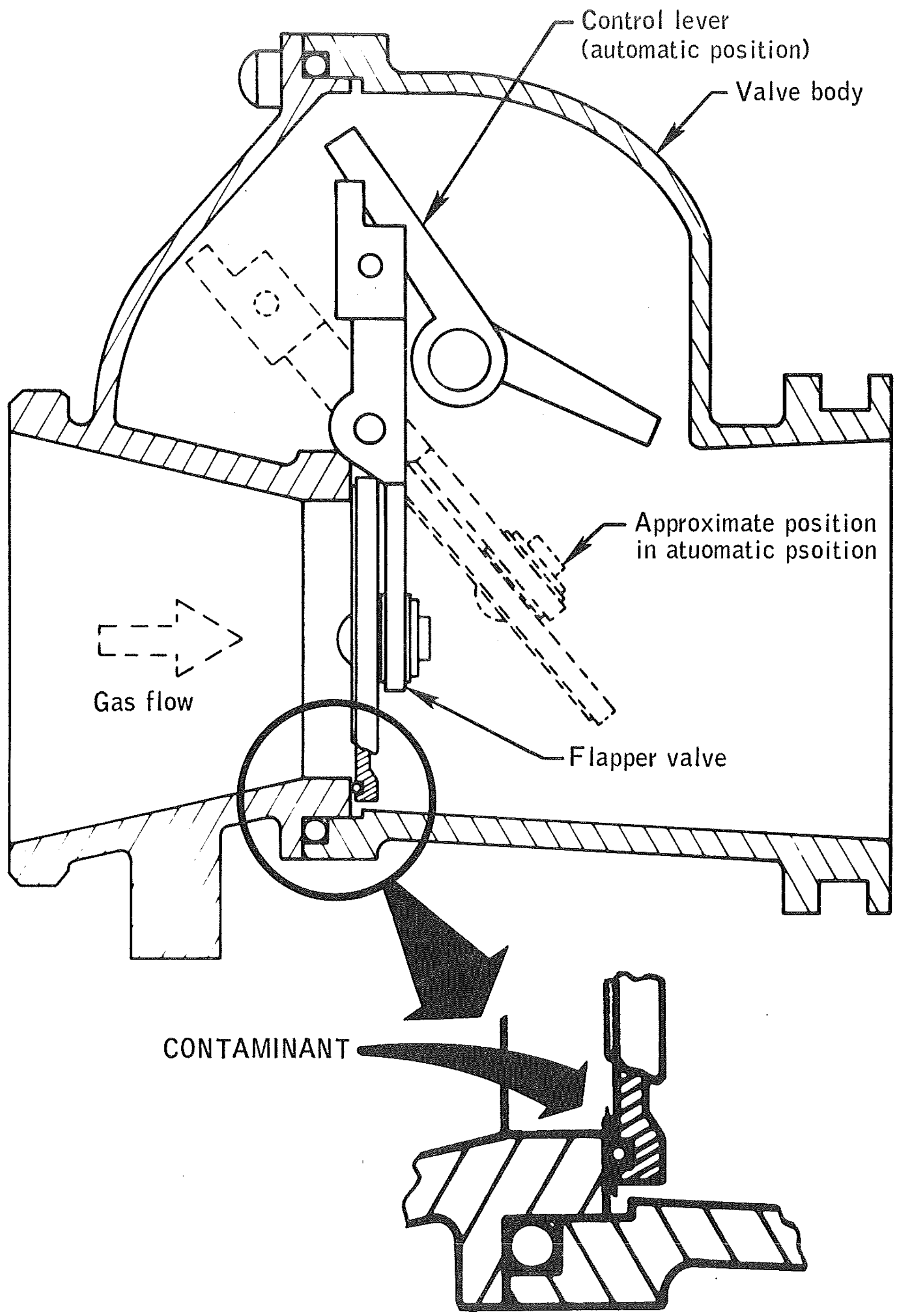


Figure 3-13.- Cabin gas return valve.

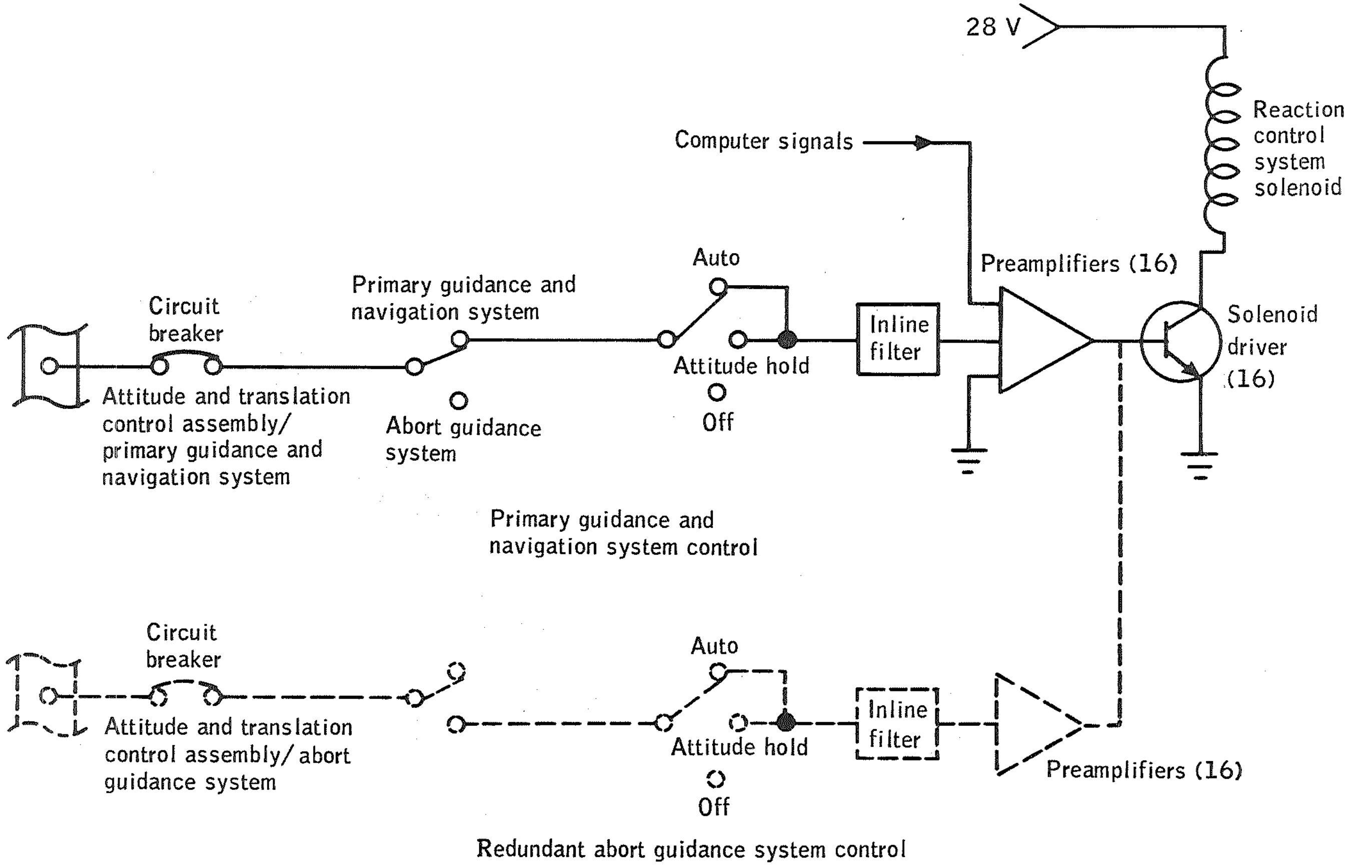


Figure 3-14.- Reaction control system control schematic.

b. The guidance control switch may have failed.

c. The primary guidance mode control switch may have failed.

d. The in-line filter could have failed.

e. An open or short-to-ground may have occurred in the spacecraft wiring or associated connectors.

f. The circuit breaker may have opened because of current loads in excess of 2 amperes.

g. The circuit breaker could have been left open by the crew.

Review of flight data and failure histories has been performed and the following conclusions made:

a. A circuit breaker on Apollo 15 failed to make electrical continuity when mechanically latched due to non-conductive contaminant between the contacts.

b. The switches in items b and c in the preceding paragraph have been X-rayed and no conductive contaminants observed. The toggle switches have no failure history of non-conductive contaminants. The guidance control switch was not changed from the position it was in during rendezvous and docking when the primary control system performed as expected.

c. Only one failure has been recorded for the in-line filter.

d. Opens or shorts-to-ground may have occurred and a short may have popped the circuit breaker, but it is unlikely that such an event occurred during this time.

e. The most probable cause of the problem is that the circuit breaker which provides 28 volts to the primary preamplifier either did not make electric contact (similar to Apollo 15) or was inadvertently left open.

Extensive changes to the flight plan and attendant procedure changes were communicated to the crew. The procedure changes were correct and were copied correctly. If the procedures for an engine hot-fire test had been required by the ground before the crew left the lunar module, an open circuit breaker or a system malfunction would have been detected. If a system malfunction had occurred, however, the deorbit maneuver could not have been executed. The deorbit maneuver can only be executed with the

primary guidance system. If a system malfunction had occured during the manned portion of the flight, adequate system redundancy is available for attitude control (fig. 3-14).

No corrective action is required.

This anomaly is closed.

3.7 ABORT GUIDANCE SYSTEM OUT-OF-PLANE VELOCITY ERROR LARGER THAN EXPECTED

Abort guidance system data during descent revealed two abnormal conditions. First, the out-of-plane component of velocity, when compared to the primary guidance system, increased to a maximum of 28 ft/sec near lunar touchdown. Second, the roll axis was misaligned after touchdown by 0.47 degrees when compared to the primary guidance system roll axis. Both of these errors are within acceptable performance limits, but they are larger than expected and are the largest seen on any Apollo flight.

Figure 3-15 is a time history of the difference between the abort guidance system and the primary guidance system X-axis attitude reference during descent. The rapid error buildup during the first 2 minutes indicates drift rates of up to 19 degrees per hour. No reason can be given for these high drift rates; however, a change in the X-axis gyro performance characteristics obviously took place. The steady drift rate of 2 degrees per hour during the last 8 minutes is believed to have been caused by a shift in a mass unbalance along the X gyro spin reference axis. Drift caused by this mass unbalance was also apparent in the data for the lunar surface calibration, due to gravity, and during ascent, due to the ascent engine thrust.

The roll axis misalignment is accounted for by the higher-than-normal drift rates during descent.

A complete review of preflight data is being conducted to verify that there was no evidence of a problem prior to launch. Hardware malfunctions within the gyro will be hypothesized and their effect on drift rates will be modeled in an effort to reconstruct the error curve in figure 3-15.

Corrective action is dependent upon completion of detailed analysis.

This anomaly is open.

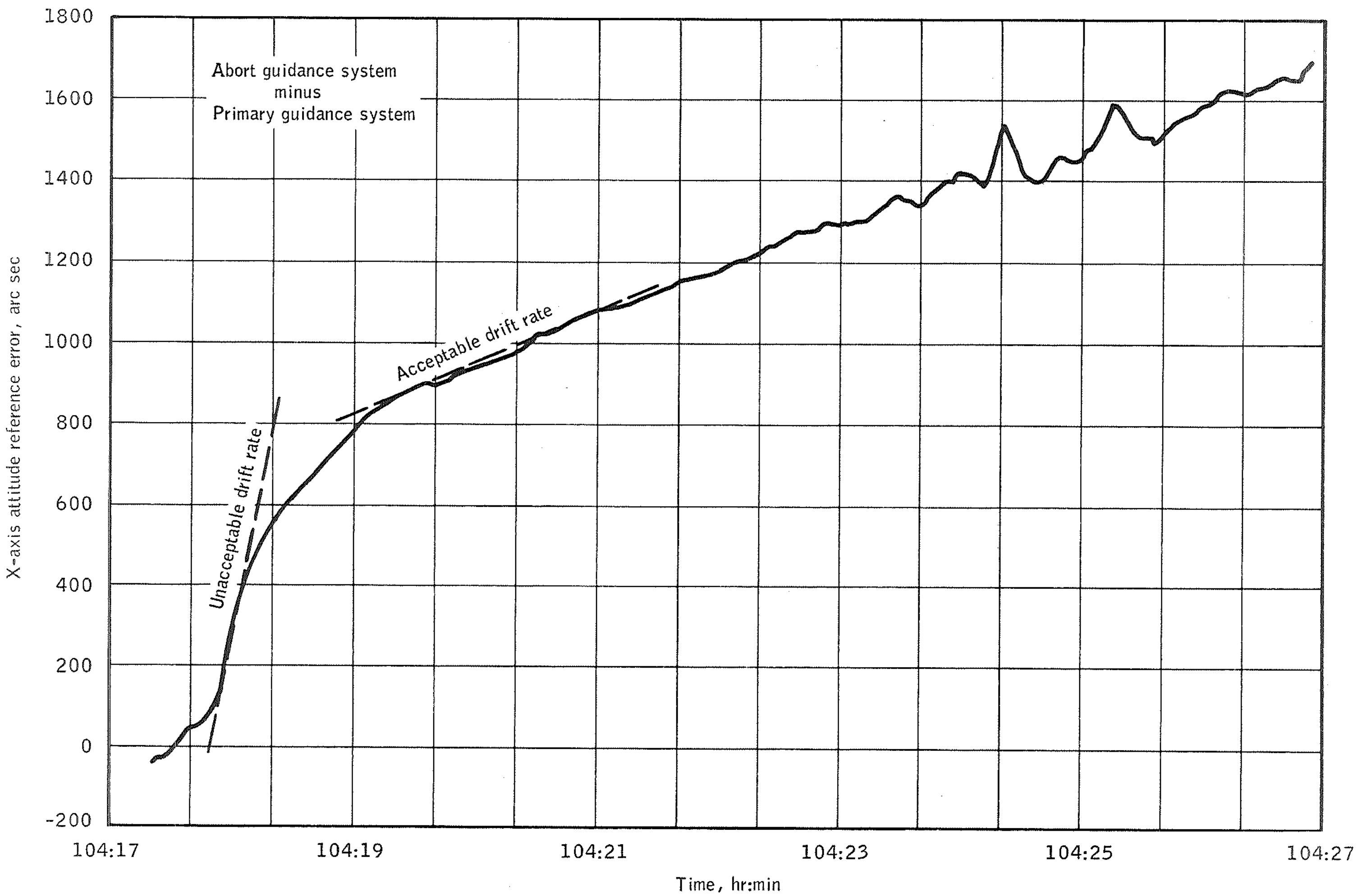


Figure 3-15.- Attitude reference error during descent.

4.0 GOVERNMENT FURNISHED EQUIPMENT ANOMALIES

4.1 MALFUNCTION OF TELEVISION CAMERA MONITOR IN COMMAND MODULE

The command module television camera monitor exhibited horizontal lines during the initial usage. This condition cleared and performance of the monitor was normal until the press conference telecast during the transearth coast phase. At that time, the monitor had the same horizontal lines reported during the initial usage, followed by a complete loss of the picture. The horizontal-hold control adjustment would not correct the horizontal line condition. The monitor was turned off for approximately 5 minutes, and then turned back on, after which the monitor picture was normal.

Since the downlink television video signal was not affected, the malfunction has been isolated to the monitor and associated cable. The system is scheduled to be tested.

This anomaly is open.

4.2 OXYGEN PURGE SYSTEM ANTENNA BROKE

Following ingress after the second extravehicular activity, the crew reported that approximately 2 inches had been broken off the tip of the antenna on the Commander's oxygen purge system (fig 4-1). The antenna had inadvertently been left unstowed while ingressing and it is believed that the antenna was broken when it struck the ascent propulsion system engine cover as the Commander was coming through the hatch.

The desirability of leaving the antennas stowed at all times is being investigated, and the feasibility of carrying a spare antenna is being considered.

This anomaly is open.

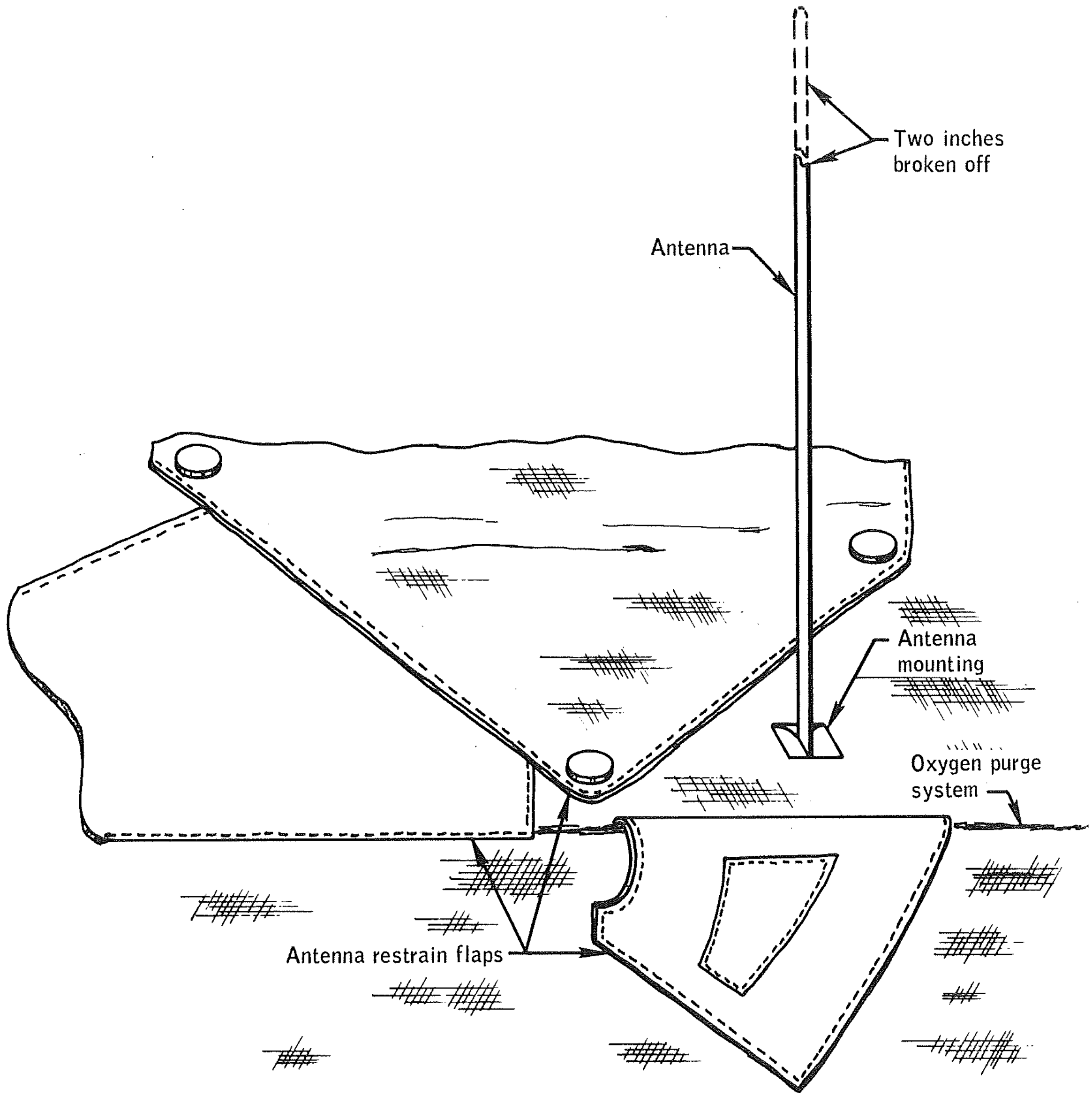


Figure 4-1.- Damaged antenna on oxygen purge system.

4.3 EXPERIMENTAL GAS/WATER SEPARATOR LEAKAGE

The experimental gas/water separator was cracked and leaking; therefore, it was not used. Postflight examination showed that the fracture occurred at the base of the threaded section of the lid (fig. 4-2) which is made of plexiglass.

The design considerations are being assessed.

This anomaly is open.

4.4 WRIST DISCONNECTS DIFFICULT TO ROTATE

After exposure to the lunar surface environment, the wrist ring disconnects on the Commander's and the Lunar Module Pilot's suits were hard to rotate to the locked position and, once locked, were very difficult to rotate out of the locked position. Lunar soil contamination and loss of lubricant are suspected as the causes of this problem.

The wrist disconnect attaches the glove to the suit sleeve (fig. 4-3) and is covered by a gauntlet on the glove. The gauntlet is loose fitting around the suit sleeve and dirt can get into the wrist disconnect. Very close tolerances between the sliding surfaces are inherent in the design. This was deliberate to make it impossible to accidentally unlock the wrist disconnects. With these design conditions, a very small amount of contamination can adversely affect the rotation of the locking mechanism.

Postflight inspection and analysis of the wrist disconnects will be performed. A glove design change which will afford better protection from contamination is under consideration.

This anomaly is open.

4.5 COMMUNICATIONS CARRIER MICROPHONE AND EARPHONE PROBLEMS

During the communications check prior to the first extravehicular activity, the microphone audio signal from the Lunar Module Pilot's headset was too weak to operate the voice-operated keying circuitry in his extravehicular communications system. This was attributed to a loose tip on one of the communications carrier headset microphone booms plus orange juice blockage of the other microphone boom (fig 4-4). Postflight analysis will be performed to determine the cause of the loose microphone boom tip. The orange juice blockage will be evaluated as a juice bag problem.

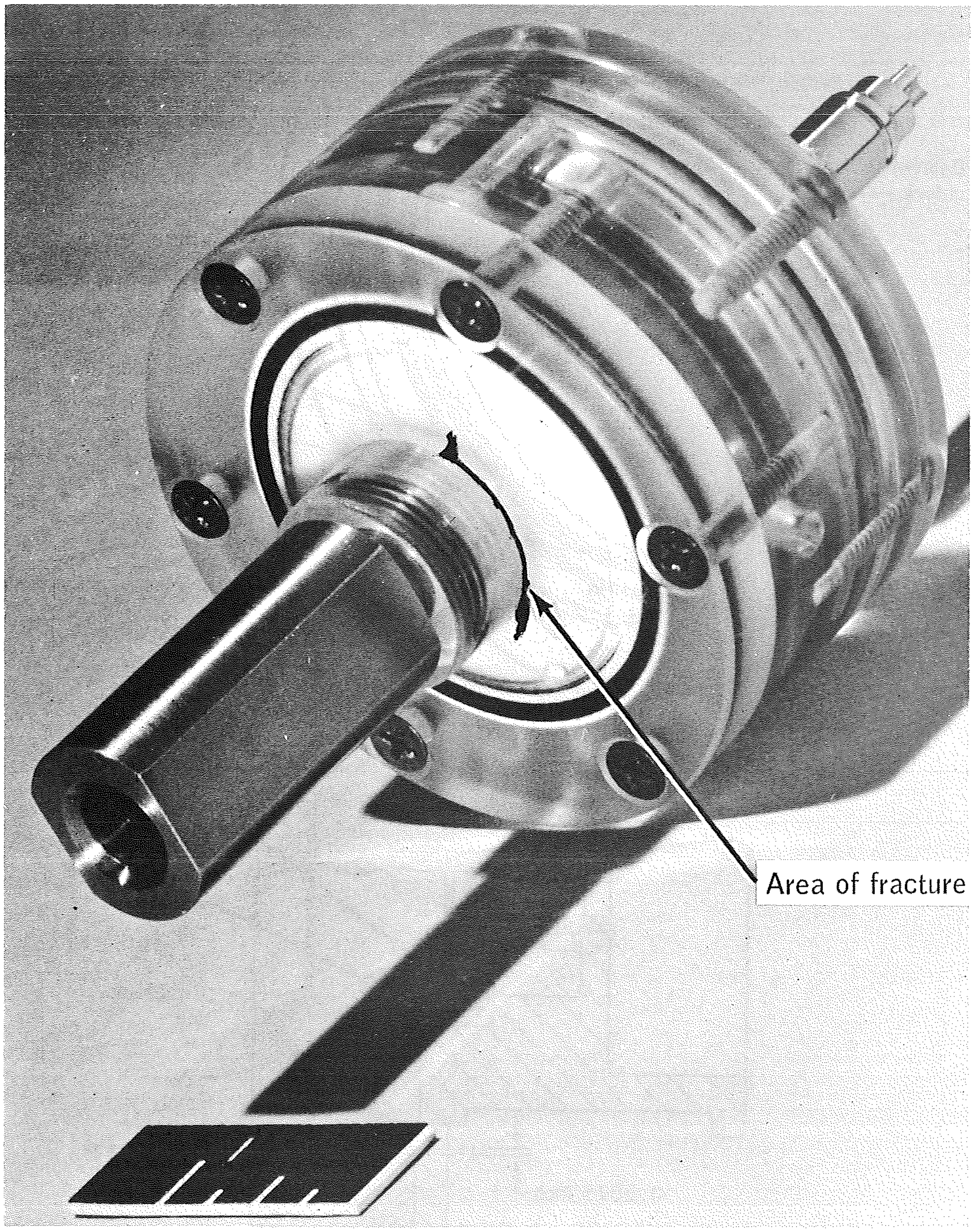
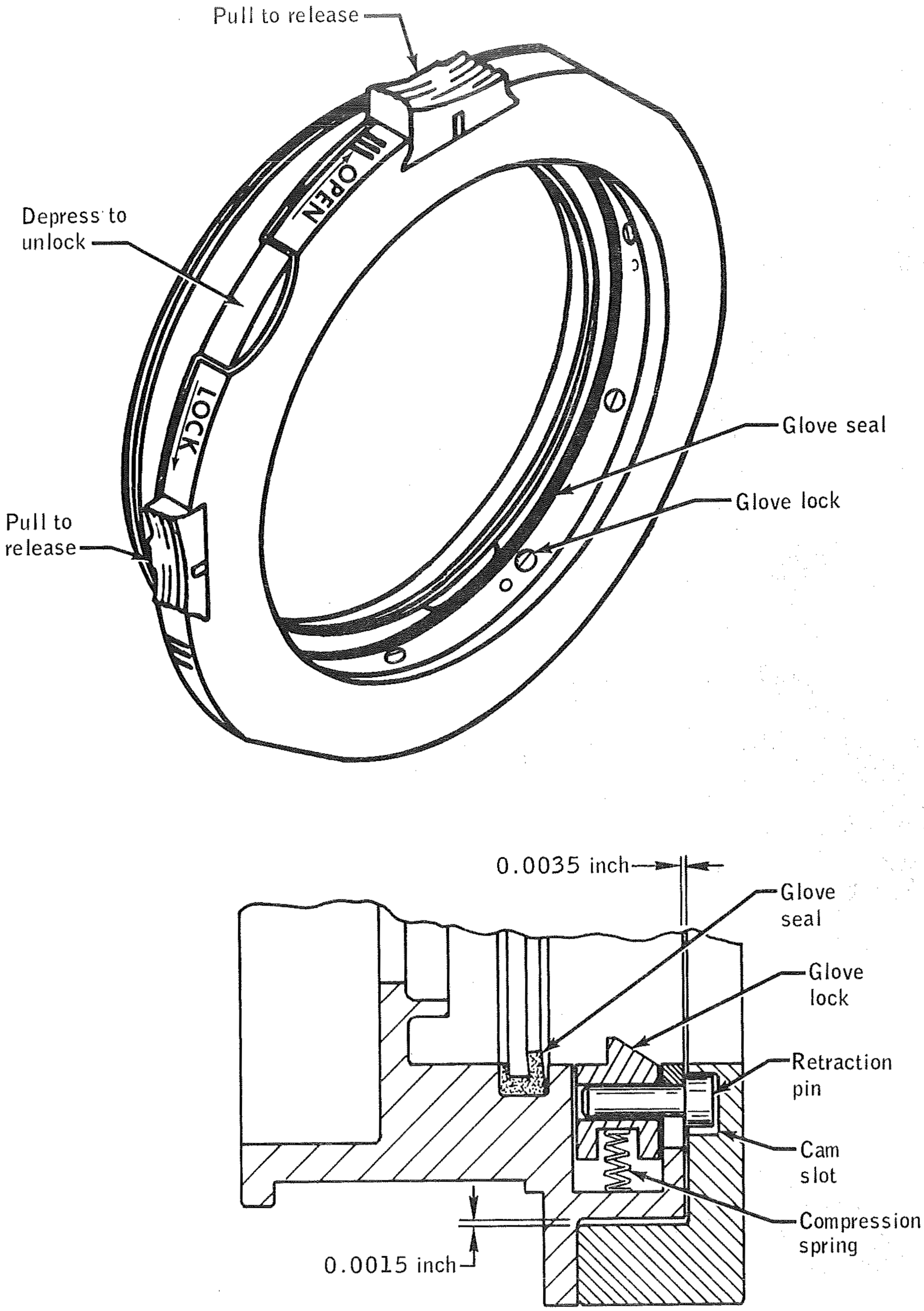


Figure 4-2.- Gas/water separator failure.



Section through ring (typical-eight places)

Figure 4-3.- Wrist disconnect.

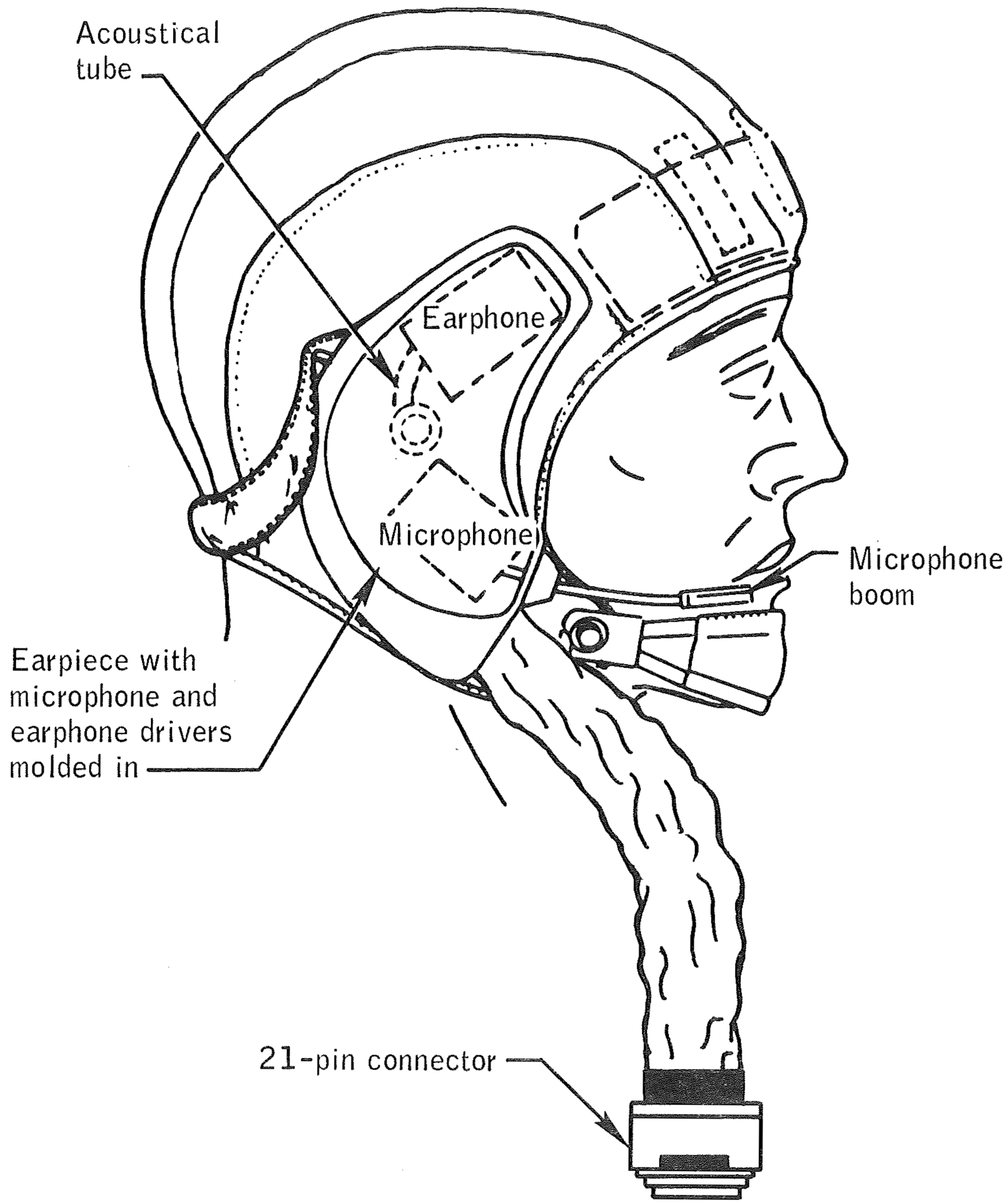


Figure 4-4.- Communications carrier.

During the transearth coast phase, the Lunar Module Pilot also experienced intermittent operation of his communications carrier headset earphone circuit. This condition was cleared by flexing the constant-wear garment harness near the 21-pin connector. Postflight tests will be made to reproduce the intermittent condition in the communications carrier umbilical and control head, the constant wear garment harness, and the communications carrier headset.

This anomaly is open.

4.6 COMMANDER'S AND LUNAR MODULE PILOT'S CUFF GAGE READINGS DIFFERENT DURING TRANSEARTH EXTRAVEHICULAR ACTIVITY

The Commander's cuff gage indicated a different pressure from that of the Lunar Module Pilot during the transearth extravehicular activity. The Commander's gage indicated approximately 3.5 psia, whereas the Lunar Module Pilot's gage indicated approximately 3.9 psia. During this period, the suit loop pressure was indicating 4.6 psia, a value which is considered to be erroneous (see sec. 2.13). The normal suit pressure should have been approximately 3.8 psia. During the suit integrity check, the cuff gages indicated approximately the same.

The cuff gage is a bellows-actuated aneroid-type device which indicates differential pressure and has an accuracy of ±0.1 psia. The gage opening to ambient pressure is protected by a screen.

Postflight tests of the gages to date show that the two suit gages are reading 3.96 and 4.03 psia compared with an actual pressure of 4.0 psia. Further analysis will be performed.

This anomaly is open.

4.7 SWOLLEN EXTRAVEHICULAR MOBILITY UNIT MAINTENANCE KIT

The extravehicular mobility unit maintenance kit stowed in the command module was swollen to about three times the size of the kit carried in the lunar module. The kits are identical in configuration. The crew reported that the visor cleaning and anti-fog pad bags were pressurized, causing the swollen condition. These bags are required to be vacuum packed to prevent swelling. Three bag assemblies are carried in each kit.

The maintenance kit will be examined.

This anomaly is open.

5.0 LUNAR SURFACE EXPERIMENT EQUIPMENT ANOMALIES

5.1 HEAT FLOW EXPERIMENT CABLE BROKE

The Commander's legs became entangled in the heat flow experiment cable near the central station and his movements resulted in the cable breaking at the connector to the central station (at the connector board solder joints); however, the central station did not move. Loss of the connection to the central station resulted in loss of the heat flow experiment.

The heat flow experiment is stowed on subpackage no. 2. Upon arrival at the Apollo lunar surface experiments package deployment site, the experiment is removed from the subpackage and the electrical ribbon cable is connected to the central station by an Astromate connector (fig. 5-1). The cable is bonded and soldered to a printed circuit board which is clamped in a connector. The cable-to-board joint is reinforced by two to four thicknesses of 0.5-mil Kapton tape.

Pull tests performed on the cable/Astromate connector configuration indicate the strength at the cable/board interface was 31 pounds. A modified joint assembly which includes a strain relief provision is under test (fig. 5-2).

This anomaly is open.

5.2 SPIKE ON MORTAR PACKAGE DID NOT DEPLOY

The lanyard pulled the release pins from three of the four spring-loaded hinged spikes and these three deployed normally. The release pin for spike 3 was bent and jammed so that it could not be pulled out.

The mortar package pallet assembly includes four 7-inch-long spikes (fig. 5-3) which deploy normal to the pallet when the release pins are removed. A spike can be deployed manually by shearing the pin; however, the crew was not adequately trained to implement this operation. When the pallet is properly positioned, the spikes are pressed into the lunar surface to provide additional stability to the platform.

Tests and analyses performed during the development of the pallet assembly indicate that three deployed spikes are adequate to provide stability. The Apollo 16 pallet was emplaced with the three spikes pressed into the lunar surface, and the crew reported the complete mortar package and pallet assembly was stable.

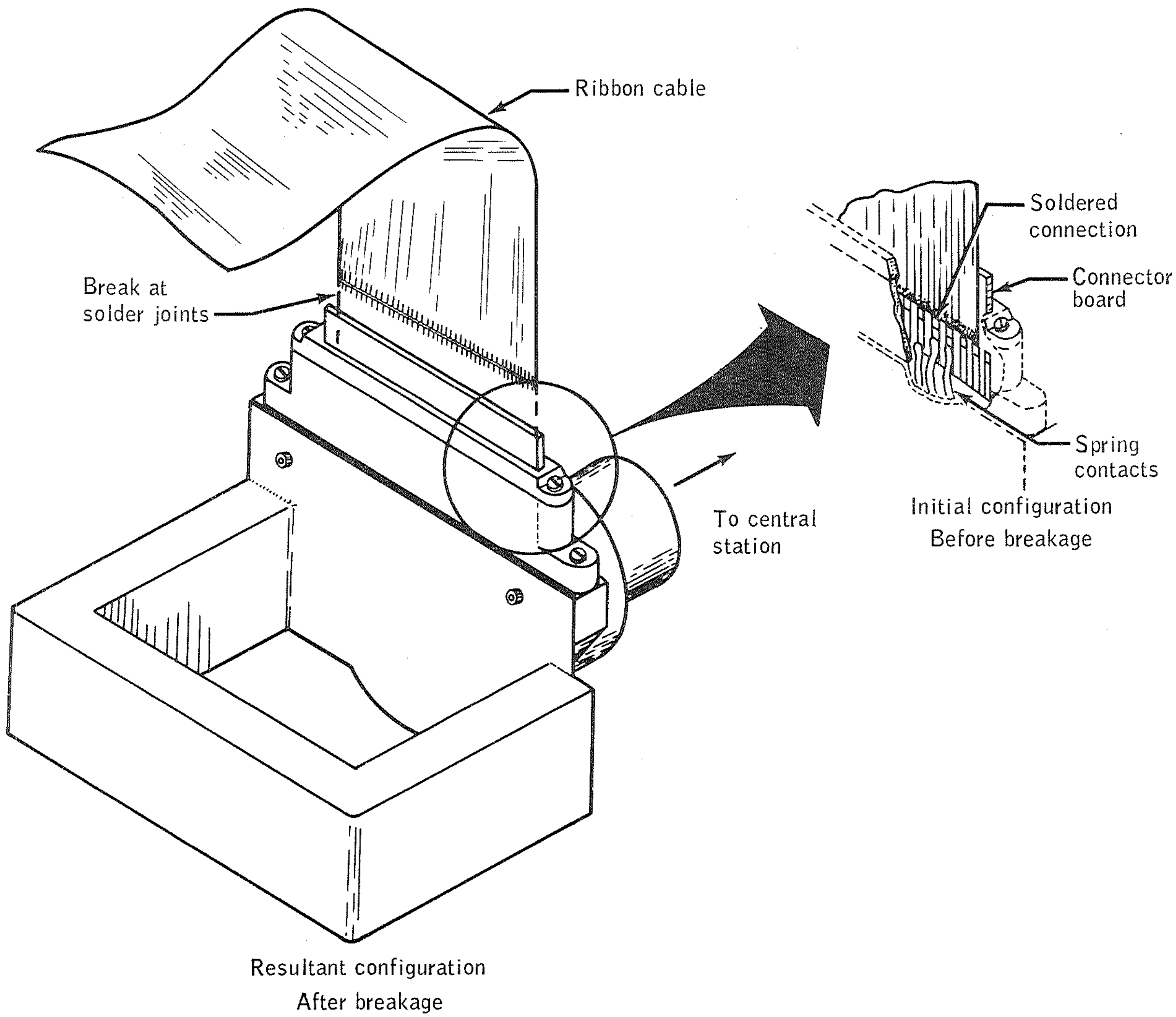


Figure 5-1.- Heat flow experiment connector.

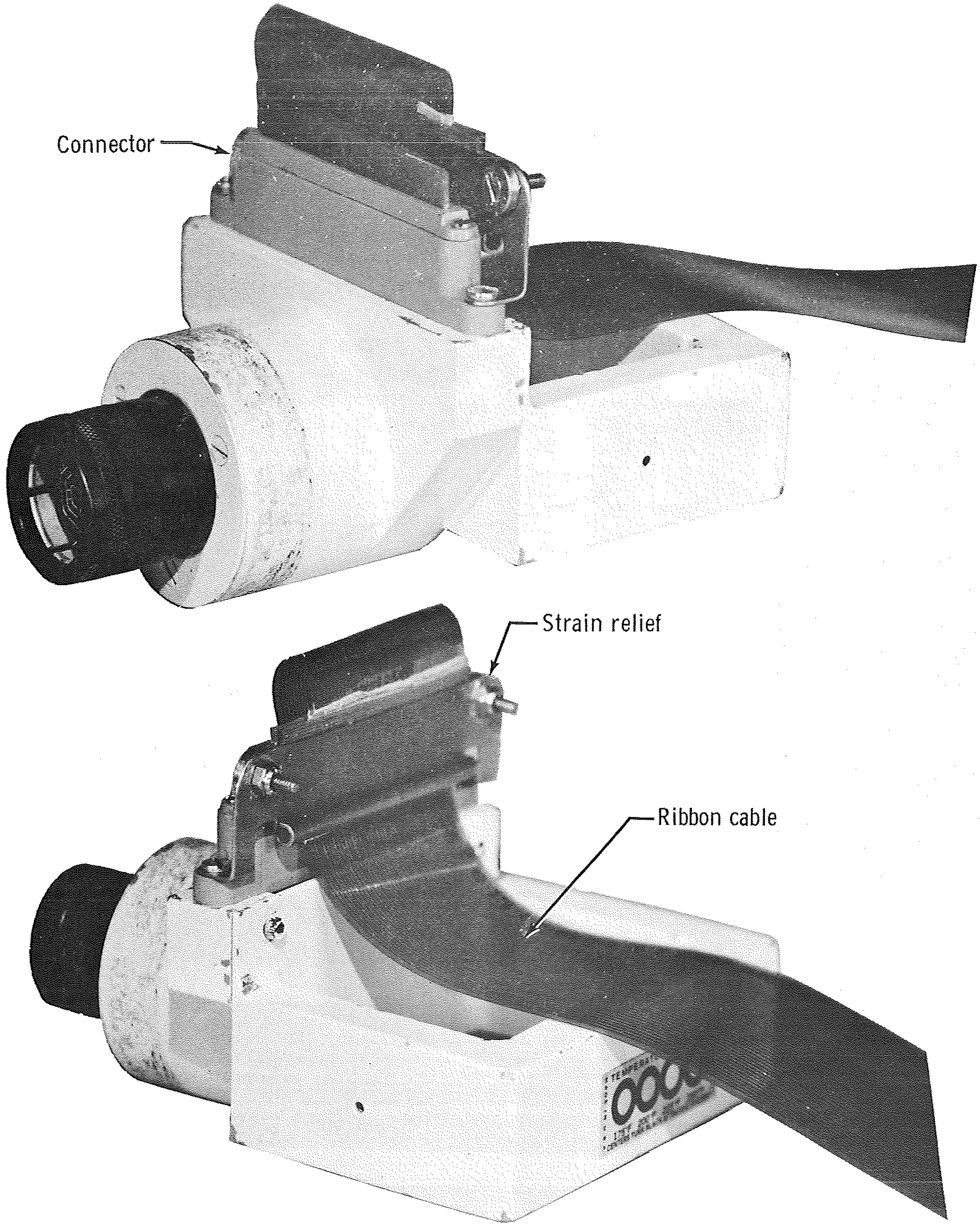


Figure 5-2. - Modified connector assembly.

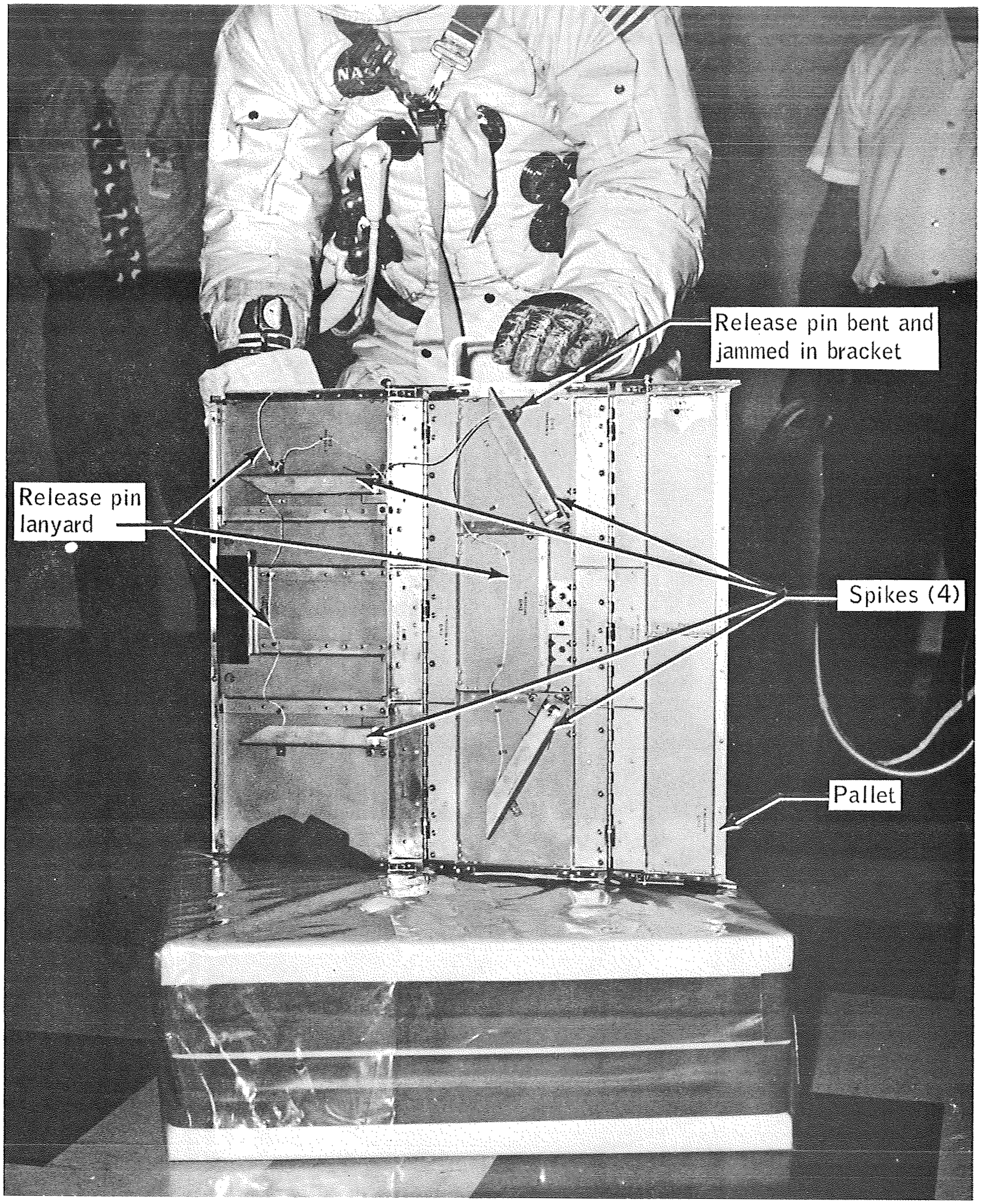


Figure 5-3.- Bottom view of active seismic experiment mortar package pallet.

No corrective action is required as the experiment is not scheduled for another mission.

This anomaly is closed.

5.3 MORTAR BOX ROLL ANGLE TELEMETRY INDICATED OFF-SCALE HIGH

The telemetry indication for the roll angle of the mortar box was off-scale high. The sensor limit is ±15 degrees. However, the bubble in the leveling device, as observed by the crew, indicated that the mortar box was less than 5 degrees off level in combined pitch and roll. The pitch angle telemetry indication was minus 2.3 degrees. Photography confirms that the alignment was satisfactory.

The sensor functions as a part of a voltage divider in an oscillator circuit. An off-scale high output could result from a failure of any component which intercepts the oscillator. At the time the roll sensor was installed on the experiment, proper operation was verified over a tilt angle of ±5 degrees.

The off-scale high measurement was the result of a component failure in the sensor circuit. The experiment is not scheduled for another mission.

This anomaly is closed.

5.4 COSMIC RAY DETECTOR PANEL 4 SHADE PARTIALLY DEPLOYED AND LANYARD BROKE

The Commander pulled the red-ring lanyard (fig. 5-4) to shift the shade in panel 4. The shade shifted only 1 inch (it should have moved about 3.5 inches), and the lanyard broke.

A movable platinum sheet covers the top half of panel 4 during translunar flight and until after the radioisotope thermoelectric generator is removed from the area during lunar surface operations. The red-ring lanyard extends from the bottom of panel 1, behind panels 1, 2, 3, and 4, and is attached to a slide assembly on the end of the movable platinum sheet. When the lanyard is pulled down, the platinum sheet is pulled up over a roller at the top of panel 4, exposing additional detectors.

Examination of the returned assembly showed that four screws in the slide frame extended too far, resulting in about 0.005-inch interference. This caused the shade to become jammed.

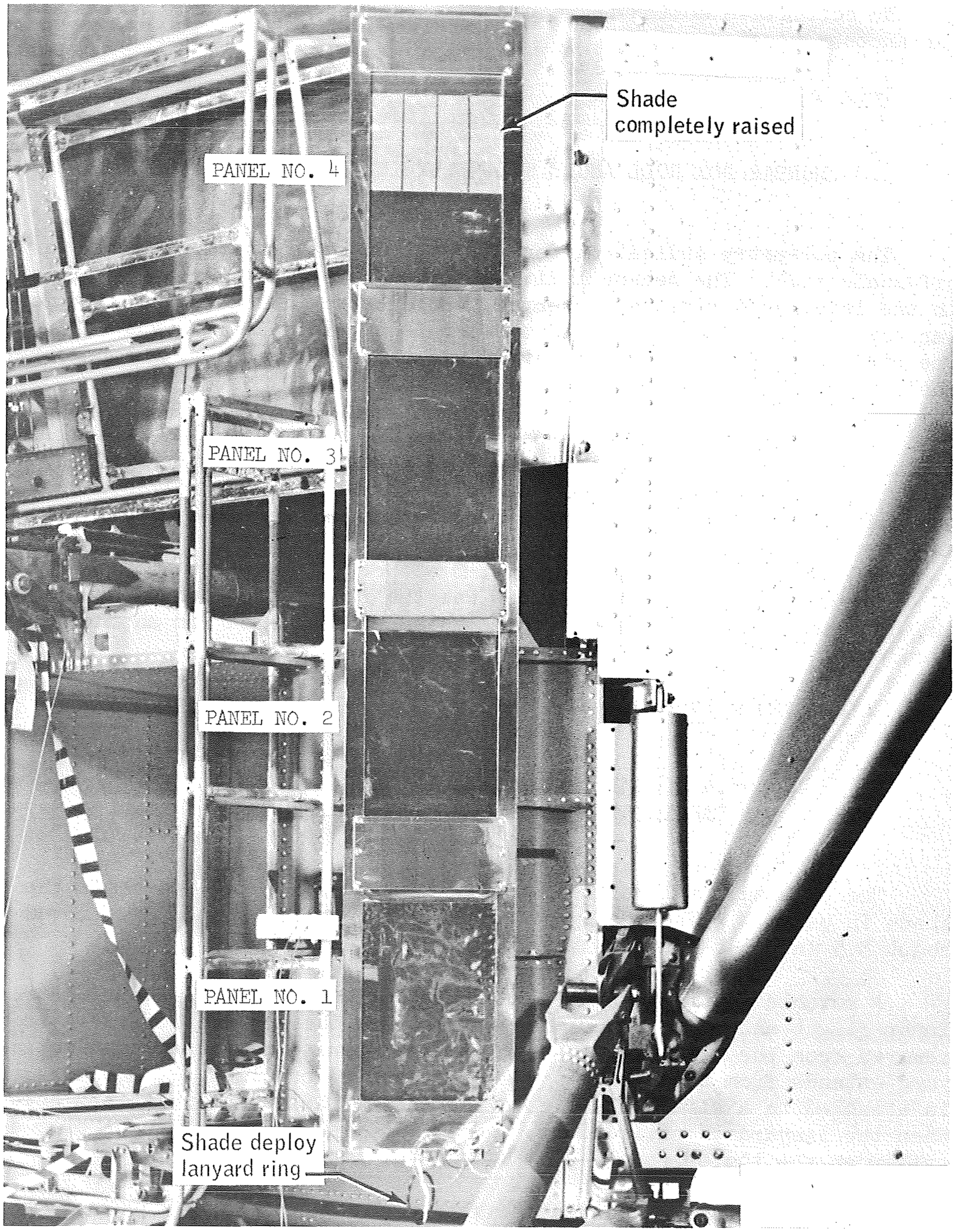


Figure 5-4.- Fully deployed cosmic ray detector experiment.

The assembly performed satisfactorily when deployed during the preflight fit and function check. However, it was necessary to refurbish the hardware after this operation, and the screw clearances may have changed as a result of the refurbishment.

This experiment is not scheduled for another mission.

This anomaly is closed.

5.5 VERTICAL STAFF OF GNOMON SEPARATED FROM LEG ASSEMBLY

When the gnomon (fig. 5-5) was being unstowed from its stowage bag during the second extravehicular activity, the leg assembly remained in the bag, and the vertical staff assembly and inner gimbal parts came out separately. The leg assembly, because of its color scale, was used during the rest of the lunar surface photographic operations.

The vertical staff/inner gimbal bearings are held in place by two pivot screws in the inner gimbal. In addition to the loctite in the threads, these screws are captive within the outer gimbal. The inner gimbal/outer gimbal bearings are held in place by two pivot screws in the outer gimbal. In case of a loctite failure, these screws could back out under vibration conditions, and allow the staff and inner gimbal to come out.

Screw retention will be assured on other gnomons.

This anomaly is open.

5.6 DOCUMENTED SAMPLE BAG DISPENSERS FELL OFF 70-MM CAMERA BRACKETS

During the geology activities on the lunar surface, the bag dispenser assemblies repeatedly fell off the brackets on the 70-mm cameras.

The dispenser is mounted on an adapter (fig. 5-6) which is inserted in the ring-sight tee-slot bracket on the camera. The adapter-to-tee slot interface is horizontal and depends on springs in the tee slot to hold the adapter in the latched position. This latching method is inadequate.

The adapter is being redesigned so as to provide a positive lock in the installed position.

This anomaly is open.

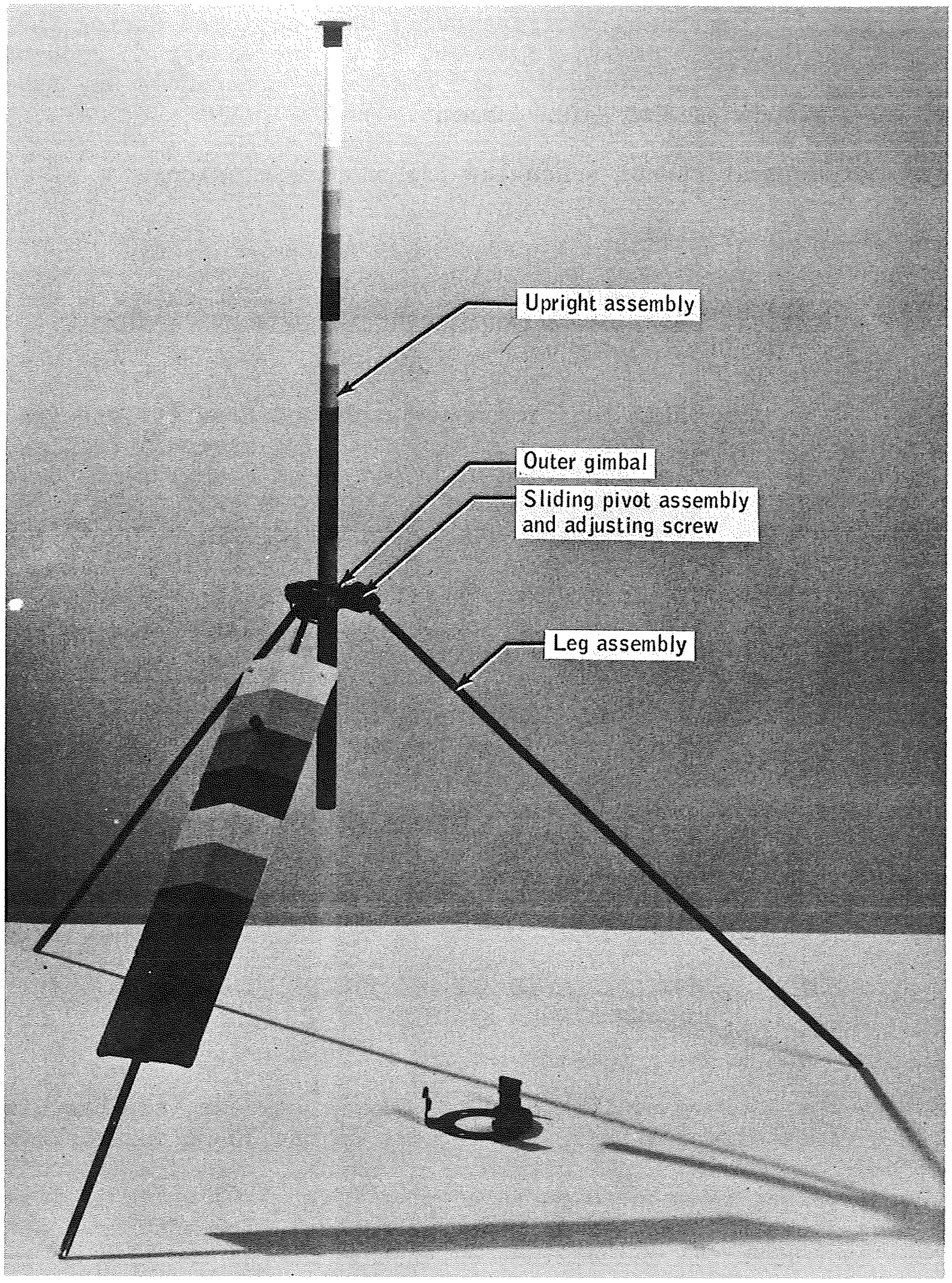


Figure 5-5.- Gnomon assembly.

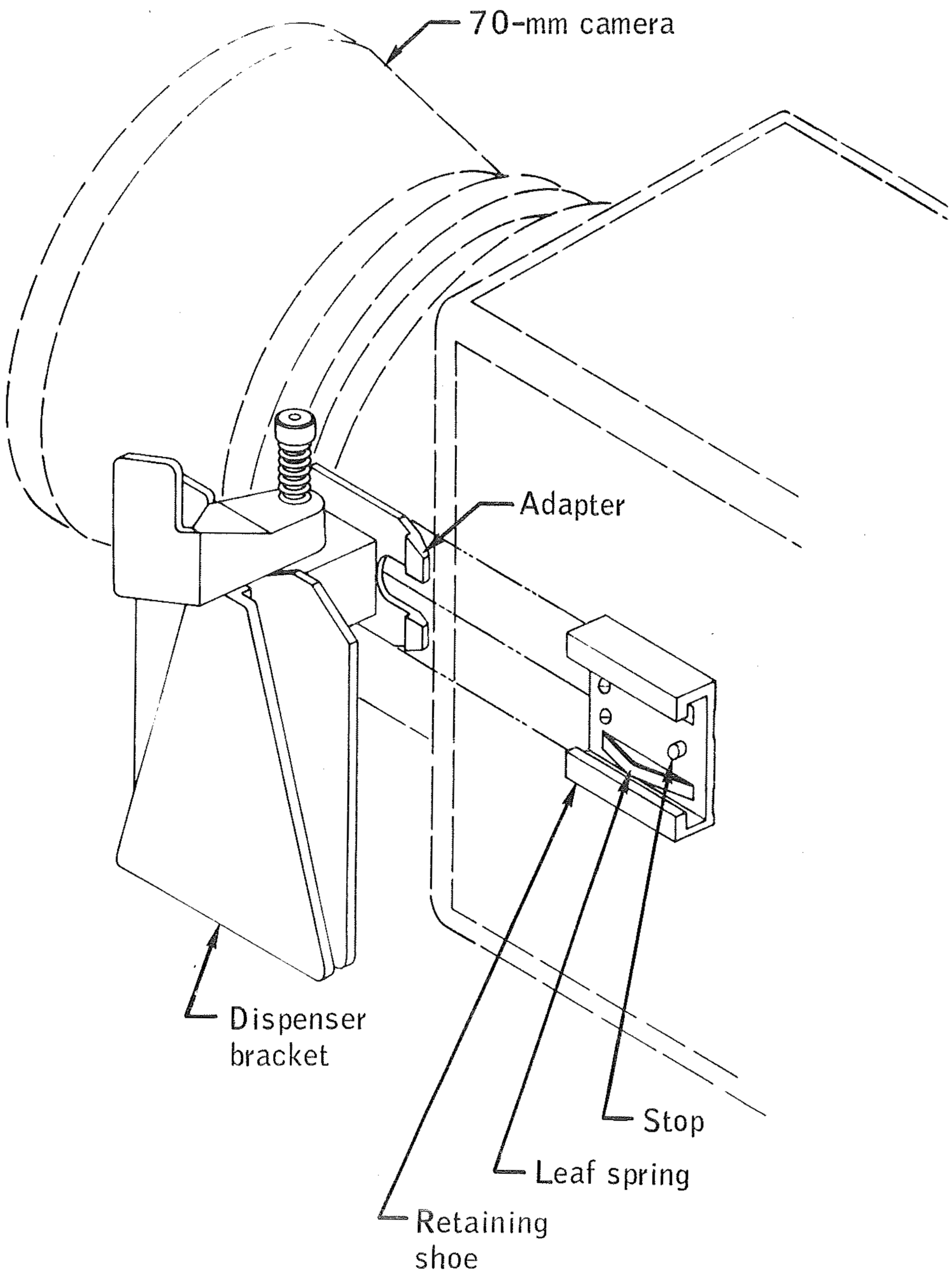


Figure 5-6.- Sample bag dispenser retention.

5.7 SCREWS AND HOLDING RING CAME OFF ONE OF THE DOCUMENTED SAMPLE BAG DISPENSERS

The crew found that one of the documented sample bag dispensers had come apart when it was unstowed. The bags were set aside and used later without the dispenser assembly.

The dispenser assembly is held together by three flat-head screws (fig. 5-7) which are retained by staking to the bracket. The use of loctite has not been permitted because of the possibility of sample contamination.

In the case of this one assembly, the staking was apparently improper and did not retain the screws. Dispensers for Apollo 17 are to be recalled for reaffirmation of adequate staking.

This anomaly is open.

5.8 SAMPLE COLLECTION BAG FELL OFF THE PORTABLE LIFE SUPPORT SYSTEM TOOL CARRIER

During lunar surface operations, the Commander's sample collection bag came loose from the portable life support system tool carrier several times and fell off once.

The sample collection bag essentially consists of a Teflon bag on a metal frame (fig. 5-8). The bag opening is covered by a Teflon lid on a hinged metal frame. Attached to the metal frame on one side of the bag, about 2 inches below the lid, is a 3/8-inch-wide stainless steel strap with offsets to accomodate the two hooks on the tool carrier. About 1 inch from the bottom of the same side is a 1-inch-wide Teflon band, sewn to the bag, with an offset loop approximately 1 inch by 5 inches to accomodate the Velcro strap from the bottom of the tool carrier. The Velcro strap, when tightened down, keeps the bag from floating or bouncing off the hooks. During the lunar roving vehicle operations, the Velcro strap sometimes loosened so that the bag could come off the hooks.

Changes under consideration include the possibility of adding a keeper to the hooks.

This anomaly is open.

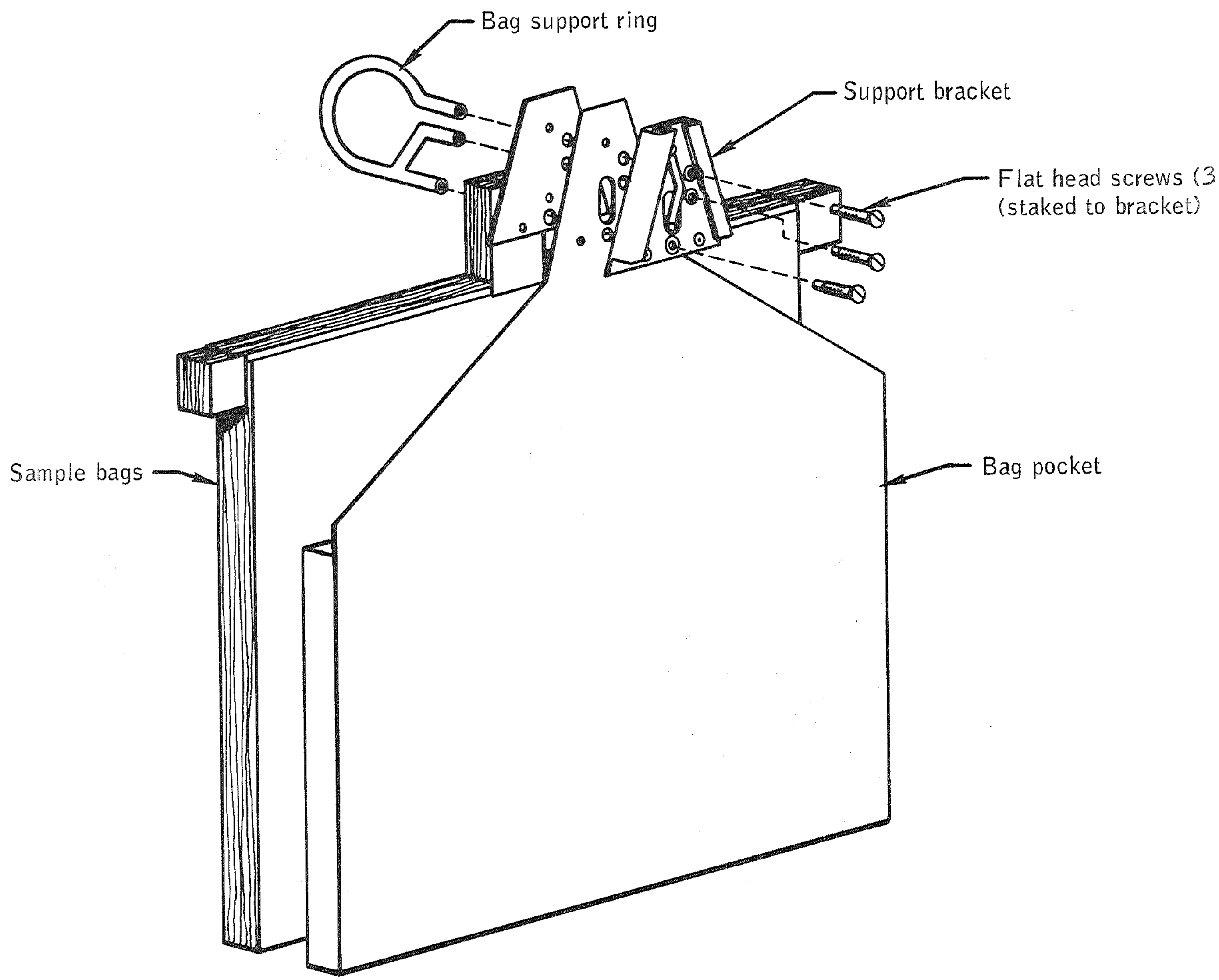


Figure 5-7.- Documented sample bag dispenser assembly.

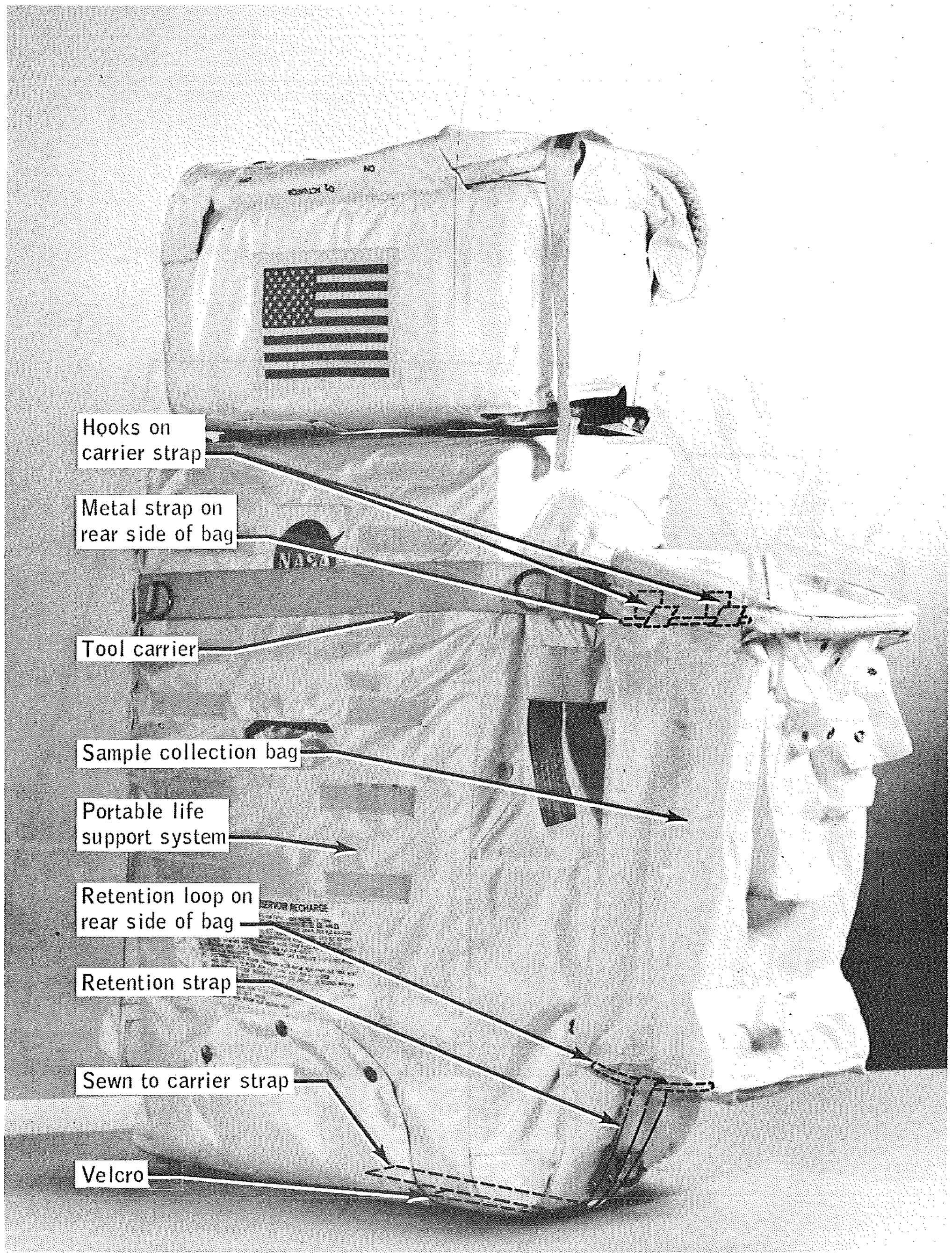


Figure 5-8.- Sample collection bag on portable life support system tool carrier.

5.9 LUNAR SURFACE ULTRAVIOLET CAMERA AZIMUTH ADJUSTMENT BECAME MORE DIFFICULT

The azimuth ring adjustment was difficult to operate and became progressively harder to move with each adjustment.

To adjust the azimuth to the proper dial reading, the camera is rotated on a 12.5-inch-diameter ball-bearing ring which is lubricated with silicone grease (fig. 5-9). The camera rests on this bearing and is retained in position by a thrust bearing (ball) on the center shaft. Neither bearing is sealed; however, the crew observed that there was no indication of dust on the bearing. Extra length of battery cable, which provides take-up between the main frame and the rotating camera, is contained in three turns in the cable trunk. The crew reported that the cable became stiff in the lunar environment, which would add to the resistance of the azimuth rotation. The camera was operated in the shade to maintain proper film and electronics temperatures.

Environmental effects on the bearings, lubricants, and cable are being investigated.

This anomaly is open.

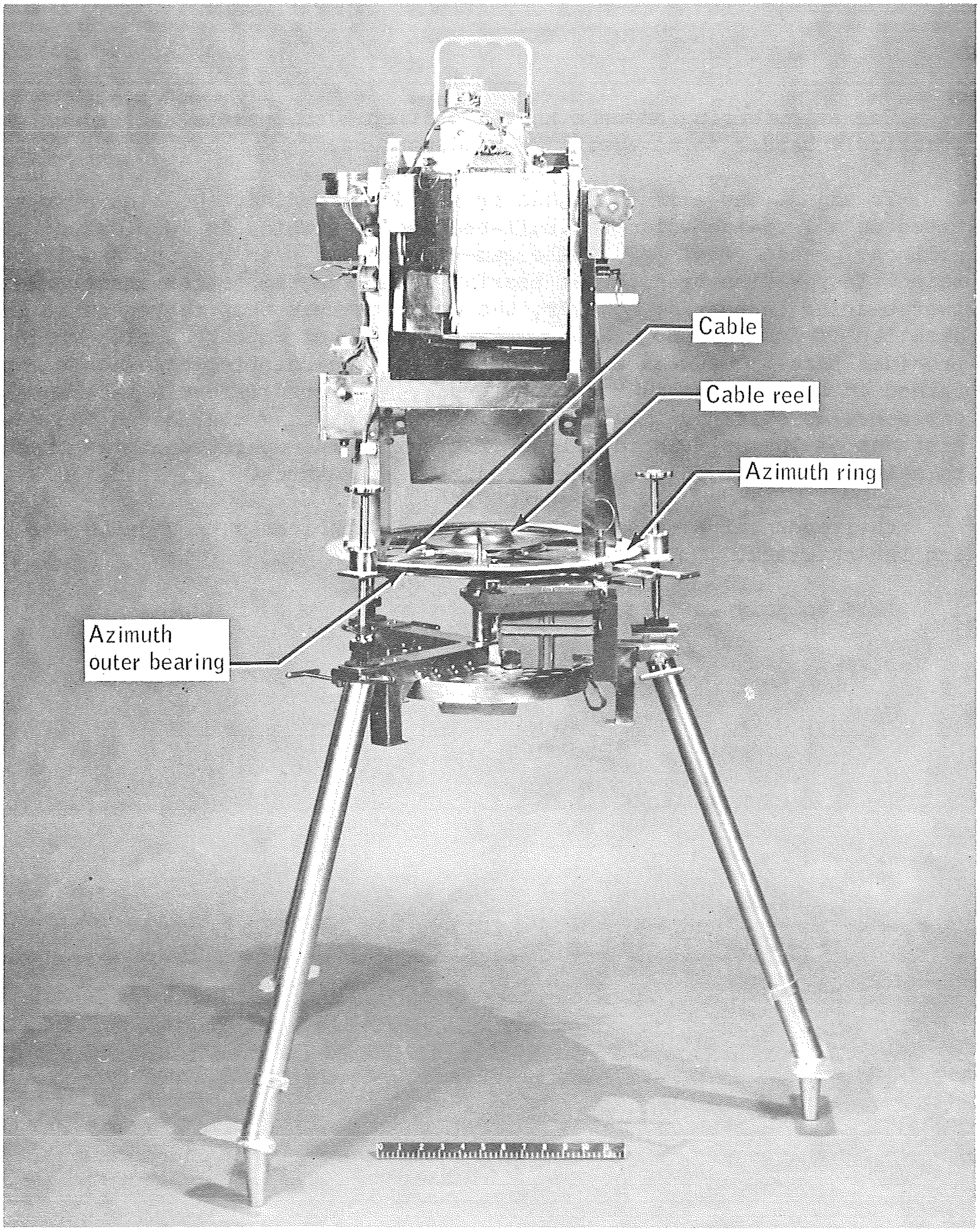


Figure 5-9.- Far ultraviolet camera.

6.0 ORBITAL EXPERIMENT EQUIPMENT ANOMALIES

6.1 MAPPING CAMERA EXTEND/RETRACT TIMES WERE ABNORMALLY LONG

The first mapping camera extension was normal, but the retraction required 2 minutes and 54 seconds. The normal time for extension or retraction is approximately 1 minute and 20 seconds. The second and third extensions and retractions required about 3 minutes, but the fourth retraction and fifth extension were both normal at 1 minute and 18 seconds. The fifth (final) retraction time was 1 minute and 34 seconds.

This anomaly is similar to the Apollo 15 problem in that the times of the first extension and retraction were normal and all subsequent times were excessive, with some in excess of 4 minutes. In addition, the Apollo 15 camera would not retract after the final (15th) deployment. Subsequent investigations did not isolate a probable cause of that anomaly but did disclose a deficiency in the technique of applying the dry film lubricant to the deployment mechanism drive screw. As a precautionary measure, the Apollo 16 drive screw was cleared of its dry film lubricant and lubricated with the silicone lubricant that is used in all bearings in the camera and deployment mechanism.

Deployment rails attached to the bottom of the camera slide through ball-bushings that are part of the deployment mechanism (fig. 6-1). Deployment of the camera is accomplished by the drive screw turning within the drive nut, which is bolted to the camera. Friction in this screw-nut drive is minimized through the use of recirculating streams of ball bearings riding between the threads of the nut and screw.

Two redundant motors, each capable of deploying the camera should the other fail, power the drive screw via a drive train consisting of clutches, gearing, and a "no-back" device that locks the deployment mechanism when power is removed from the motors. The motors, clutches, and most of the gearing are contained within the gear box. Two clutches are used, one with each motor. Their design is such that, should a motor seize, its clutch will decouple from the drive train. The clutches contain bearings that are lubricated with a mixture of silicone oil and grease. It is possible that this lubricant may be transferring, by migration or outgassing, to the friction surfaces of the clutches.

A re-evaluation of the entire deployment mechanism design is in progress. Tests to determine the effect of lubricant contamination of clutch friction surfaces will be performed.

This anomaly is open.

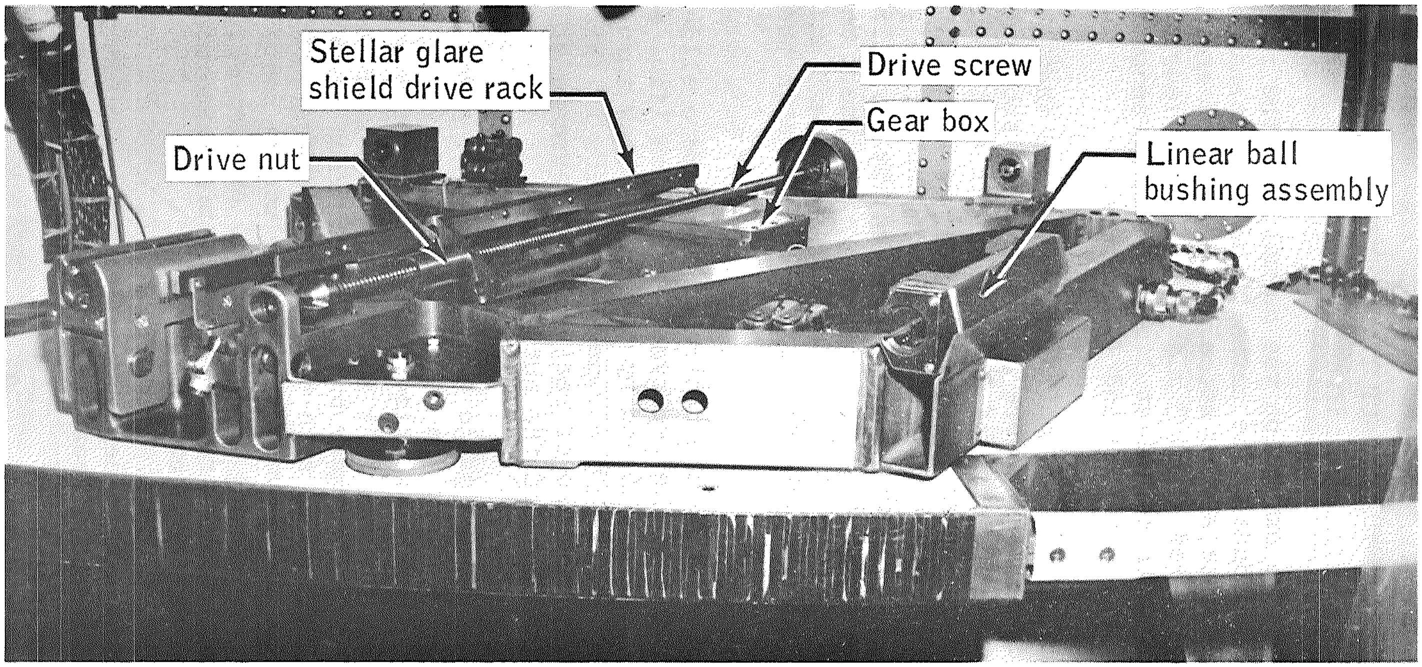


Figure 6-1.- Mapping camera deployment mechanism.

6.2 MAPPING CAMERA STELLAR GLARE SHIELD FAILED TO RETRACT

The Command Module Pilot observed that the glare shield for the stellar camera was jammed against the handrail paralleling the scientific instrument module bay (fig. 6-2) during the transearth extravehicular activity. In the 16-mm photography of the extravehicular activity, the outermost edge of the folding tip of the glare shield is visible above the handrail, thereby substantiating the Pilot's observation that the glare shield was in the fully-extended position as shown in figure 6-2.

Scientific instrument module bay photographs taken from the lunar module indicate that the glare shield was properly retracted at rendezvous; therefore, the failure to retract occurred at the third, fourth, or fifth (last) camera retraction. A preliminary review of the stellar film indicates that this anomaly did not affect the stellar photography.

Glare shield deployment is accomplished by a rack-and-pinion mechanism that is connected by gearing to a rack on the camera deployment mechanism (fig. 6-1). As the camera is deployed from the scientific instrument module bay, the glare shield is automatically extended outward from the stellar camera. The glare shield reaches full extension or retraction before the camera is completely deployed or retracted; therefore, there is a clutch in the glare shield drive train which slips after the glare shield is fully extended or retracted. The slippage continues until the camera deployment or retraction is complete. The clutch also functions as a safety device to preclude interference with camera deployment in the event of a glare shield extension malfunction.

Since the clutch is similar in design to the clutches used in the mapping camera deployment mechanism, an investigation is being performed to determine effects of migration of lubricant to the clutch friction surfaces.

This anomaly is open.

6.3 LASER ALTIMETER OUTPUT POWER DEGRADED

The laser altimeter performed normally during its first operating period - 41 minutes during lunar revolutions 3 and 4. Evidence of laser degradation began to appear early in the second operating period (revolution 16). The pulse forming network voltage controller began stepping to voltage position 1 (step 0 is the initial setting), and the laser output power,

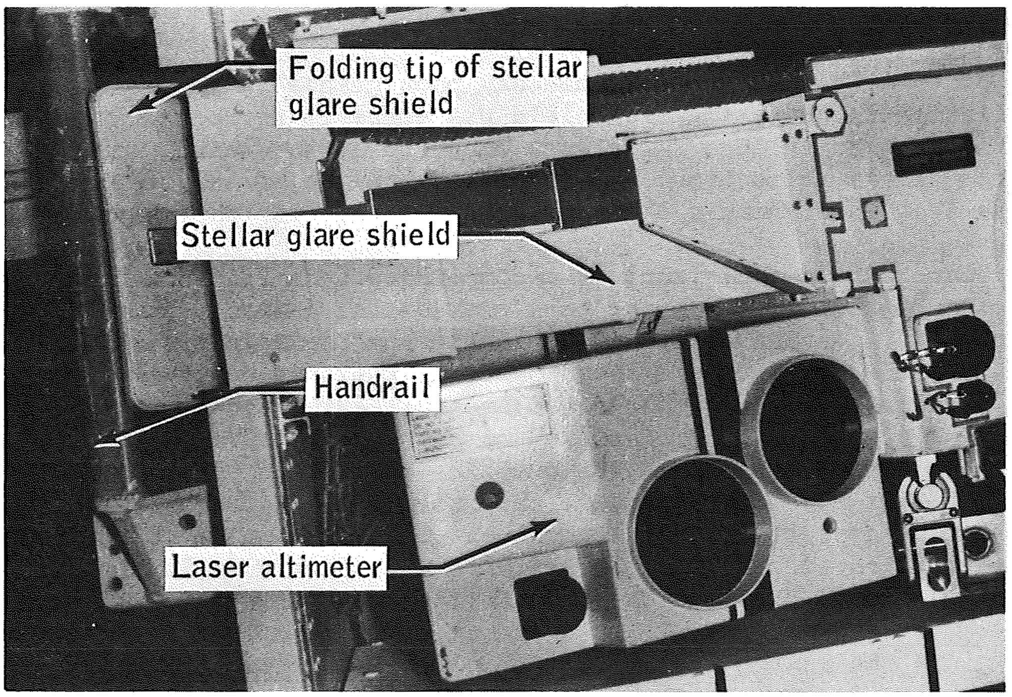


Figure 6-2.- Mockup photograph of glare shield handrail.

when in step 0, was occasionally too low for ranging - as manifested by altitude readings of zero meters and an overflowing range counter. For this period, altitude data validity was about 68 percent; i.e., approximately 32 percent of the data was "zero".

The degradation rate appeared to be rather constant during the next three operating periods, with voltage step 4 appearing near the end of the fifth period (revolution 47). During the sixth operating period (revolution 60), the degradation appeared to accelerate and the pulse forming network voltage controller reached step 5, the highest available step, by the end of this run. In step 5, maximum voltage is delivered to the flashlamps which "pump" the laser ruby. During the last scheduled operating period (revolution 63), laser output degraded to the point that no further altitude data were obtained.

There were approximately 2400 laser operations (firings) during the mission. The laser output appeared to degrade more rapidly than during the Apollo 15 mission (figure 6-3) but was compensated by the action of the pulse forming network voltage controller added for this purpose as a result of the Apollo 15 anomaly. The pulse forming network consists of a laser power status sensor (photodiode) located inside the laser module and associated circuitry to increase the output of the pulse forming network power supply if the sensed laser power is lower than a pre-established threshold.

Compensation was sufficient until the last 40 minutes of the final scheduled operating period. After this time, laser output was apparently too low for ranging against the sunlit lunar surface. One consequence of the pulse forming network controller operation was some loss of altitude data on alternate laser firings; therefore, data validity was only about 70 percent during most of this mission. Had the controller not been added to this unit, the amount of valid data obtained would have been significantly less. An estimate of the data that would have been obtained is shown in figure 6-3. The controller operates to increase input power to the laser if the laser output on the preceding operation was low, and to decrease input if the output was high. It appears that the threshold may have been low, or the voltage steps too large, so that inadequate output was obtained on the lower step, resulting in loss of altitude data. This was particularly true on illuminated portions of the lunar surface where background radiance decreases receiver sensitivity.

The laser module was processed in the same way, with the new type of bearing and other refinements, as the module in the delta qualification test. This test was conducted in a vacuum for 220 hours with operation on a simulated mission timeline for 50.5 hours. Total operations exceeded 9000 firings and the voltage controller had reached step 4 at the end of the test. Similar performance was expected of the Apollo 16 laser. The

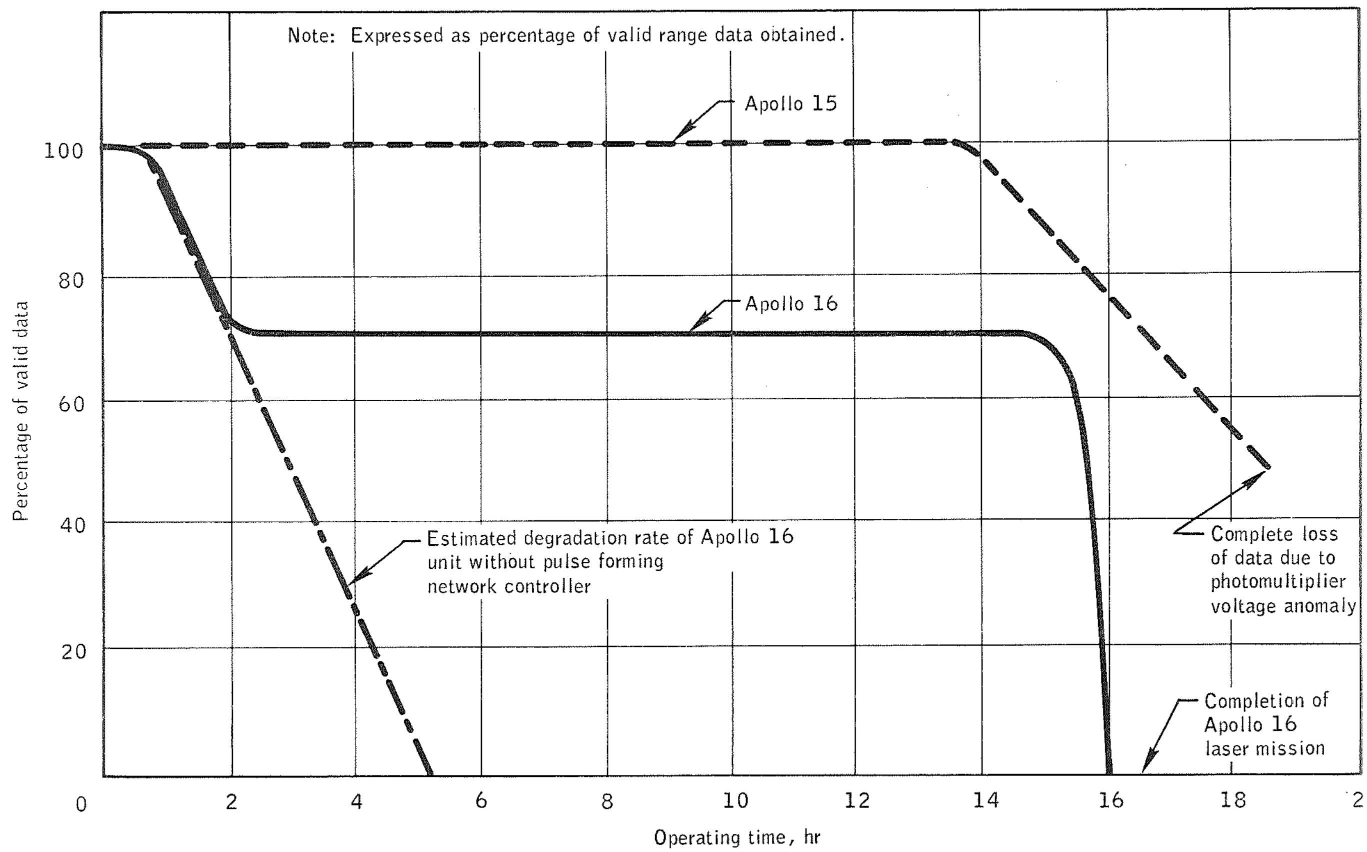


Figure 6-3.- Laser degradation versus operating time.

number of prelaunch operations of the Apollo 16 laser was purposely restricted to conserve life; therefore, the rapid degradation could not be discerned preflight.

The plots of data validity versus operating time in figure 6-3 are approximate in that the effects of laser module temperature and lunar surface illumination have been averaged out. During the first 11 hours of laser operation during Apollo 15, the command and service module was in a highly eccentric orbit which, due to the range limitations of the laser altimeter, caused the loss of approximately 50 percent of the altitude data. Since the data loss was not due to a hardware malfunction, the curve for Apollo 15 is drawn at 100 percent during this period.

Contamination within the laser module may have caused the degradation. Investigations are in progress to determine if the amount of lubricant in the Q-switch bearings can be reduced and to determine the feasibility of confining the Q-switch motor-rotor coupling area so that motor lubricant would not be transferred to the optical surfaces. The Q-switch consists of a prism rotating at approximately 22 000 revolutions per minute so that the optical alignment required for lasing is maintained for an extremely short period, thereby resulting in compression of the laser energy into a high-amplitude pulse about 13 microseconds in duration. Investigations will also be conducted to determine if flashlamp degradation and surface finish of the ruby rod are possible failure causes. Additionally, an investigation is in progress to determine means for improving the operation of the laser power status sensor.

This anomaly is open.

6.4 PANORAMIC CAMERA AUTOMATIC EXPOSURE CONTROL INDICATED LOW LIGHT LEVELS

The panoramic camera contains a light sensor and associated circuitry that determines the film exposure. As the scene luminance decreases, the output from the sensor decreases; the resulting signal delivered to the slit width servo causes the exposure slit to widen, thereby automatically increasing the film exposure. The slit width and the light sensor output are both telemetered (fig. 6-4).

Throughout the mission, telemetry indicated that the sensor output was too low, resulting in a slit width that was too wide. Consequently, an overexposure of the film occurred. Based upon preflight brightness calculations and real-time mapping camera exposure data, the degree of overexposure was estimated to be approximately 1 1/2 f-stops. As a consequence, the film processing was modified to compensate for this degree of overexposure. A preliminary review of the processed film indicates that the photography was not compromised.

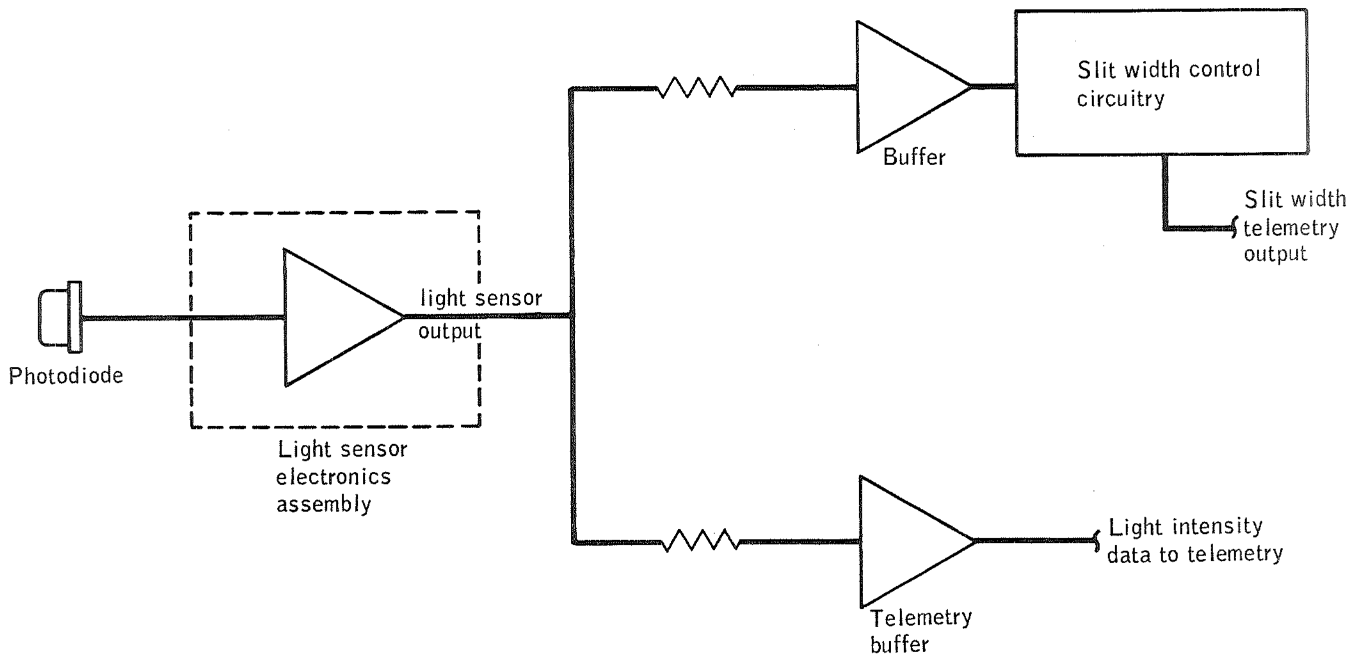


Figure 6-4.- Block diagram of panaramic camera automatic exposure control

Preliminary analysis indicates that the problem was, most likely, the result of a failure within the light sensor or its circuitry (fig. 6-5). Breadboard tests are being conducted in an effort to isolate the failure and prelaunch data are being reviewed to determine if the problem could have occurred prior to flight.

This anomaly is open.

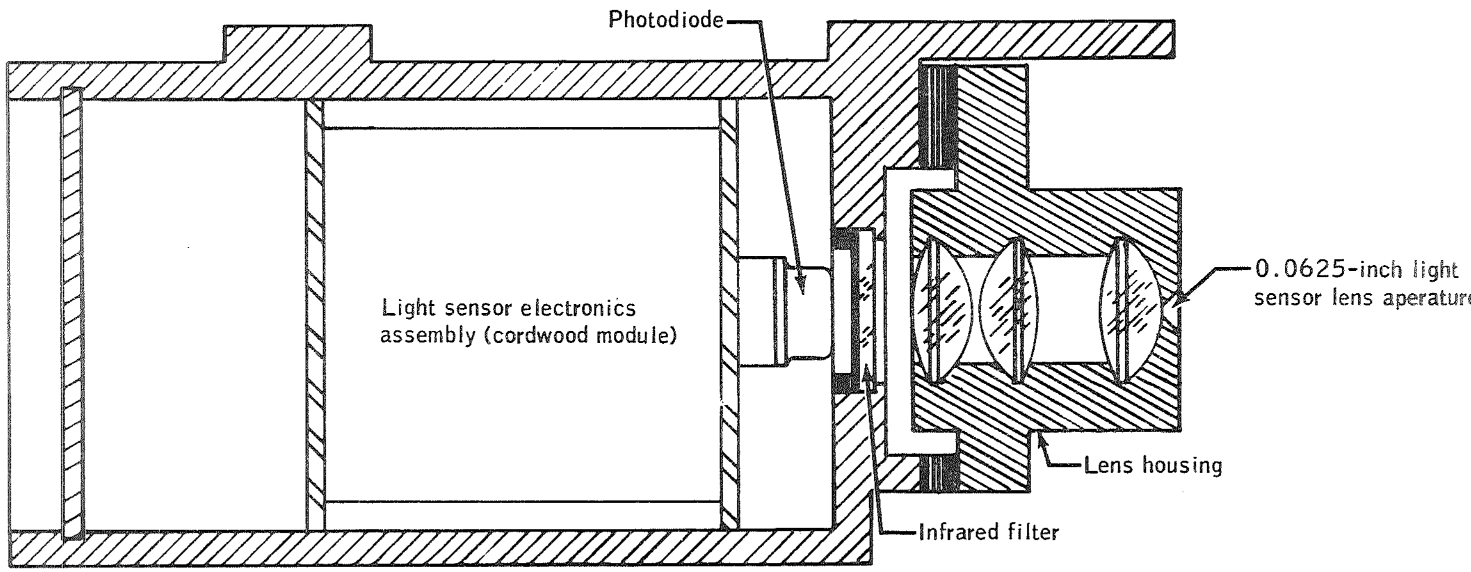


Figure 6-5.- Panoramic camera light sensor assembly.

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